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NACA

RESEARCH MEMORANDUM

NACA RESEARCH ON COMBUSTORS FOR AIRCRAFT GAS TURBINES

I - EFFECT OF OPERATING VARIABLES ON STEADY-STATE PERFORMANCE

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SUMMARY

Systematic research on current aircraft gas-turbine combustors was conducted to determine and to generalize for this type of combustion system the effect on performance of operating variables, of geometric or design variables, and of fuel variables. The effect of operating variables on steady-state performance of turbojet combustors is presented; the results described pertain to liquid-fuel turbojet combustors of the type in general use at present and operating on the fuels for which they were designed - gasoline or kerosene. Trends depicting the effect of inlet-air pressure, temperature, and velocity and fuel-air ratio on performance characteristics, such as combustion efficiency, maximum temperature rise attainable, pressure loss, and combustor-outlet temperature distribution are described for a number of combustors. These trends are further discussed as they effect significant changes in the turbojet engine, such as altitude operational limits, specific fuel consumption, thrust, acceleration, and turbine life.

The combustion efficiency decreases if combustor-inlet pressure p or temperature T is decreased, or if combustor velocity V is increased. The parameter pT/V plotted against combustion efficiency generalizes a large amount of combustor data adequately. Because of the effects of the combustor-inlet pressure and temperature, combustion efficiency decreases with increased altitude or decreased engine speed.

The maximum temperature rise attainable from a given combustor also decreases if combustor-inlet pressure or temperature is decreased, or if combustor reference velocity is increased. As a result, for any aircraft gas turbine, at each engine speed an altitude exists above which the engine will not operate because of a limit in temperature rise available from the combustor. These same factors similarly impose a limit on the temperature rise available for acceleration at altitudes below the operational limit just described.

According to examination of data on various combustors, more than one-half of the usual 4- to 6-percent loss in the combustor of the inlet total pressure is usually due to aerodynamic drag in the combustor. 1393

The average deviation of the combustor-outlet temperature from the mean is unaffected by inlet-air conditions as long as combustion is stable and vapor lock does not occur in the fuel-injection system. The temperature deviation increases with the outlet temperature. Thus the average deviation from the mean combustor-outlet temperature is generally the greatest at high-altitude, full-engine-speed conditions where both high outlet temperatures and inefficient and unstable combustion are encountered.

INTRODUCTION

The NACA Lewis laboratory is engaged in a program of combustion research to aid in achieving the most effective utilization of combustion for flight. This program involves fundamental studies directed at understanding basic combustion processes and systematic investigations directed at application of combustion processes to combustors for aircraft engines. The fundamental studies provide the background information for future developments and aid in the interpretation and the understanding of combustor research. The NACA systematic research on aircraft gas-turbine combustors is being conducted to determine and to generalize for this type of combustion system the effect on performance of operating variables, of geometric or design variables, and of fuel variables. It is the intent of this study to review and to summarize the principal results from NACA research on current gas-turbine combustors.

This paper is limited to a discussion of the effect of operating variables on steady-state combustor performance (reference 1). The effects of design and fuel variables, as well as the transient-state problems of ignition and acceleration and any analytical solution of gas-turbine-combustor performance that proves possible, are considered outside the scope of this paper. Basically, the operating variables are inlet-air pressure, temperature, velocity, and fuel-air ratio. The primary performance characteristics observed are combustion efficiency, maximum temperature rise attainable, pressure loss, and combustor-outlet temperature distribution. The trends observed are further discussed as effects significant to the turbojet engine, such as altitude operational limits, specific fuel consumption, thrust, acceleration, and turbine life. Although most of the trends were observed with turbojet combustors, they should also be directly applicable to turbine-propeller combustors.

The data as shown and discussed are selected as representative of the many data available from investigations that have been conducted with six different can-type combustors, with five different annular combustors, and with numerous modifications to several of these annular combustors. The fuels used in the work reported here were the fuels for which the combustors were designed; that is, gasoline- and kerosene-type fuels. The data in this report are intended to be illustrative, to indicate the manner in which present aircraft gas-turbine combustors operate, and to aid research on and further development of such combustors.

AIRCRAFT GAS-TURBINE COMBUSTOR

The combustor for the aircraft turbine engine must efficiently convert the chemical energy in fuel and air to thermal energy for doing useful work. The special requirements that an aircraft turbine engine be flexible in operation and of light weight and low frontal area impose certain unique demands on the aircraft gas-turbine combustor. Not only must the combustor burn fuel continuously and efficiently, but a large amount of fuel must be burned in a small space and in a short time. The pressure loss in the combustor must be kept low in order to avoid excessive loss in engine-cycle efficiency; the distribution of temperature at the combustor outlet must be free of "hot spots" and be of a pattern to provide maximum turbine blade life; starting and acceleration of the engine must be reliable; combustion should not cause carbon deposits or soot; and, of course, the combustion equipment must be lightweight and durable. With all these requirements, the combustor must operate over a wide range of conditions. For example, in a typical turbojet combustor, inlet-air pressure may be as high as 75 pounds per square inch absolute or as low as 2 pounds per square inch absolute; inlet-air temperature may vary from approximately 400° to -65° F; and air mass flow may vary over more than a thirtyfold range. Combustor velocities (that is, reference velocity based on the state of the air at the inlet, the air mass flow, and the maximum cross-sectional area of the combustor) are currently of the order of 30 to 200 feet per second. Water injection in the engine for thrust augmentation, as well as climatic conditions, imposes wide variations of humidity at the combustor inlet. Temperature-rise requirements may vary from 50° or 100° to 2000° F or more, representing theoretical fuel-air ratios of from about 0.001 to 0.025, a range that is clearly outside the range of inflammability limits determined in static tests for hydrocarbons and air at a pressure of 1 atmosphere.

Early application of combustion to the aircraft gas turbine was described by Frank Whittle, British pioneer in jet propulsion, in 1945 (reference 2). The development of a combustor using atomized spray injection of the fuel is attributed by Whittle to the assistance of Lubbock. Since this initial discussion by Whittle, Shepherd (reference 3), Lloyd (reference 4), Watson and Clarke (reference 5), Mock (reference 6), Nerad (reference 7), and Way (reference 8) have also described the further development of aircraft gas-turbine combustors and have discussed their observations of the performance of such combustors.

In order to aid in describing the general combustion technique that has been evolved to meet the requirements of aircraft gas turbines for the widely variant conditions mentioned in the "Introduction," several pertinent facts are noted here:

(1) The inflammability limit of fuel in air may be expected to set an apparent limit on the combustor that must be overcome. In figure 1 are shown the inflammability limits for a 100-octane gasoline in air and the required operating range for a typical turbojet combustor with the over-all fuel concentration indicated for the combustor. The data show that the over-all fuel-air ratio of the combustor is at all times too lean for combustion. Provision must be made in the combustor for the achievement of fuel-air ratios within the inflammability limits. The inflammability limits progressively narrow as pressure is reduced below about 4 pounds per square inch, and below 0.68 pound per square inch absolute no ignition at all is possible, which indicates an increasingly difficult combustion problem at low pressures with an absolute minimum pressure for combustion existing. Research by other investigators has shown that this minimum ignition pressure for hydrocarbons in air is nearly independent of initial temperature of the mixture, at least over the temperature range -50° to 300° F.

(2) The temperatures required at various pressures to cause spontaneous ignition of a hexane-air mixture near stoichiometric proportions are shown in figure 2. This curve is typical of those found for many hydrocarbons. The temperature required for igniting the mixture rises appreciably as pressure is decreased, which implies that efficient combustion will be more difficult to initiate and to maintain at low pressure.

(3) The energy required of an electric spark to ignite a propane-air mixture as a function of gas velocity at three low pressures is shown in figure 3. The mixture composition is approximately that requiring minimum energy. These data not only demonstrate further

that it will be increasingly difficult to maintain continuous initiation of combustion as pressure is decreased, but also indicate that the energy needed for ignition increases as the time of exposure of the combustible charge to the ignition source decreases.

The ignition delays for two fuels added as liquid droplets to air at a pressure of 1 atmosphere and at various temperatures are shown in figure 4. If typical reference velocities of current turbojet combustors are considered, a fuel droplet passing straight through the combustor would have about 0.02 second to ignite and to burn completely. From figures 2 to 4, it is apparent that not only must a high-temperature zone for the ignition of fuel be maintained, but provision must be made for allowing sufficient time for combustion to proceed to completion.

In figure 5 are shown the uniform flame velocity in a quiescent mixture of hexane in air at 1 atmosphere and the velocity required to blow out a flame of premixed gasoline vapor and air seated on a plate punched with 1/4-inch holes to provide 17.2-percent-open area. The flame speed of gasoline does not differ appreciably from that of hexane. Because of the sheltered zones behind the plate and of turbulence, the velocities required for blow-out are much higher than the flame velocities measured in quiescent mixtures. Also shown in figure 5 is the over-all velocity and fuel-air-ratio requirement for the typical turbojet combustor. This figure shows the necessity of having turbulence and "sheltered" areas in the combustor to provide time for the combustion process to proceed to completion. A consideration of figures 5 and 1 again emphasizes the necessity of providing a zone for combustion where mixture ratios near stoichiometric will exist.

Figures 1 to 5 are in no way exhaustive of the fundamentals of combustion, but are presented only to show in a general way the nature of the combustion problems for the aircraft gas-turbine engines. In brief, figures 1 to 5 show that satisfactory combustion requires fuel-air ratios in the range near stoichiometric, that fuel and air must reach pressures and temperatures above definite limiting values for combustion to occur, and that sufficient time must be allowed for combustion to be initiated and completed. From a consideration of typical combustor requirements as imposed by the engine and as noted in figures 1, 2, and 5, it is apparent that the over-all operating conditions for gas-turbine combustors do not coincide with the conditions requisite for combustion and that a special design technique is required if combustion is to occur at all.

The basic gas-turbine-combustor designs that are currently in general use are shown in figure 6, which illustrates three alternative arrangements of combustor for aircraft gas-turbine engines. In figures 6(a) and 6(b), several tubular or can-type combustors are symmetrically arranged around the engine between the compressor and the turbine, and in figure 6(c) an annular combustor occupies the same space, but is obviously a single combustor. The can-type combustor may be either a reverse-flow type (fig. 6(a)) or a straight-through type (fig. 6(b)); the straight-through type is in more general use than the reverse-flow type. As the arrows in figure 6 show, air passes through holes distributed in the walls of a flame tube, or liner, that is concentric within the outer casing of the combustor. Ordinarily a hydrocarbon fuel such as gasoline or kerosene is admitted at the upstream end of the combustor. The fuel may be sprayed as an atomized liquid (atomizing combustor), may be externally vaporized and admitted, or may be vaporized inside the combustor prior to being mixed with air and burned (vaporizing combustors); liquid spray is currently the most widely used method. The fuel is ignited and burns with the air in the upstream end of the combustor, that is, the primary air. The fuel-air ratios in this zone of combustion are obviously richer than the over-all operating fuel-air ratios (figs. 1 and 5); low initial velocities, turbulence, and air-flow reversals in this zone insure that fresh charge is brought to ignition temperatures (fig. 2), and provide time for the combustion to be initiated and to occur (figs. 3 to 5). As the burned or burning gases pass downstream, more air, that is, the secondary air, enters the flame tube and dilutes and cools the gases to temperatures tolerable to the turbine. Thus the requirements for combustion depicted in figures 1 to 5 are essentially met. Satisfactory combustion will depend on fulfillment of these conditions and burning of all the fuel before quenching by the combustor walls and by the secondary-air terminates combustion. Typical heat-release rates (Btu/(hr)(cu ft)(atm)) of turbojet combustors that have been developed according to the principles described are about 100 times that of a natural-draft gas burner and about 20 times that of the modern marine oil-firing burner.

APPARATUS AND INSTRUMENTATION

Apparatus

A diagram of a typical apparatus for the investigation of gas-turbine-combustor performance is shown in figure 7. The combustor is connected to the laboratory air-supply and exhaust systems. Both room-temperature air, with pressures ranging up to 60 pounds per square inch absolute, and refrigerated air, with temperatures ranging down to

-70° F, are available. The inlet-air temperature is regulated by means of either an electric or combustion heater. Both atmospheric and altitude exhaust, with pressures ranging down to 4 inches of mercury absolute, are available. The air flow and pressure in the combustor are regulated by remote-controlled valves located upstream and downstream of the combustor.

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A calming chamber is located upstream of the combustor to give a uniform velocity distribution in the combustor-inlet duct. The ducts at the inlet and the outlet of the combustor are fabricated to simulate the air passages in the design engine. A primary cooler, consisting of pressure-atomizing spray nozzles that inject water into the exhaust gases, is located upstream of the exhaust-regulating valve. A secondary cooler of similar design is installed farther downstream to further cool the exhaust gases when they are passed into the altitude-exhaust system.

Observation windows are provided at several locations to permit visual observation of the combustion.

Instrumentation

The principal instrument stations, as shown in figure 7, are station 1 at the combustor inlet, station 2 at the combustor outlet, and station 3 located some distance downstream of the combustor. At station 1 are located iron-constantan thermocouples and pressure probes to determine the inlet-air temperature and total pressure, respectively. The thermocouple junctions and the pressure probes are placed at centers of equal areas in the duct, with one thermocouple junction and one pressure probe for approximately 8 square inches of duct area. Also at station 1 are located wall static-pressure taps, generally at four stations equally spaced about the periphery of the duct, and two or more stream static-pressure probes.

At station 2 are located chromel-alumel thermocouples and pressure probes to determine the combustor-outlet temperature and total pressure. The thermocouple junctions and total-pressure probes are placed at centers of equal areas in the duct, with one thermocouple junction for approximately 2 square inches of duct area and one pressure probe for approximately 4 square inches of duct area. The thermocouples and the total-pressure probes were constructed in the form of multiple-point rakes. A diagram of a typical six-point thermocouple rake is shown in figure 8(a); the insert shows some details of construction. A diagram of a typical four-point total-pressure rake is shown in figure 8(b). Also at station 2 are wall static-pressure taps similar in number and location to those at station 1.

Two or more multiple-shielded chromel-alumel thermocouples of the type shown in figure 8(c) are generally inserted through packing glands at station 3 so that horizontal and vertical traverses can be made to determine average temperatures and thereby check the readings obtained 1393
at station 2.

The thermocouples are connected through multiple switches to calibrated self-balancing potentiometers. The fuel flow is metered by calibrated rotameters. The air flow is metered by a square-edge orifice installed and calibrated according to A.S.M.E. specifications. The pressure taps are connected to a bank of manometers and these manometers are photographed to obtain data records.

TECHNIQUES AND PROCEDURES

Methods and Accuracy

The combustor-inlet total pressure and temperature are taken as the arithmetic mean values of the individual readings, which generally vary less than ±3 percent from the mean values at this station. The combustor-outlet temperature and total pressure are also taken as the arithmetic mean values of the individual readings at station 2. The individual readings at station 2 vary considerably from the mean values, as will be subsequently discussed.

Thermocouple indications are taken as true values of total temperature with no correction for radiation, stagnation, or conduction along the thermocouple leads. The radiation effects are minimized by insulating the outside of the duct at station 2, thereby causing the duct wall to attain a temperature only about 100° F below the mean gas-stream temperature. Bare-wire thermocouples are used at station 2 because a large number of readings are necessary to obtain a reasonably reliable mean value of the combustor-outlet temperature; shielded thermocouples would give too much blockage of the duct area. The technique is described and discussed in reference 16.

The accuracy of the combustor-outlet temperatures obtained from the bare-wire thermocouple indications was checked in three ways, according to reference 16: (1) Two multiple-shielded thermocouples of the type shown in figure 8(c) were placed just downstream of two of the bare-wire thermocouples at station 2 in a typical combustor. With a mean gas temperature of 1000° F, the readings of both types of thermocouple fluctuated as much as ±50° F with time. Readings of both types of thermocouple were taken almost simultaneously, and the shielded thermocouples agreed with the bare-wire thermocouples within 10° F with neither type thermocouple reading consistently higher than the other. (2) In a typical test rig, comparisons were made between the mean combustor-outlet temperature obtained from the bare-wire thermocouples at station 2 and the mean temperature obtained from the traversing shielded thermocouples at station 3. After correcting for the temperature change due to heat loss through the duct walls between stations 2 and 3, the mean temperatures at the two stations were found to agree within 10° F with neither station giving a value consistently higher than the other. (3) With several combustors, as the combustor-inlet conditions were made more favorable, the combustion efficiency progressively increased and asymptotically approached 100 percent.

Since reference 16 was written (May, 1948), bare-wire thermocouples have been compared with thermocouples having radiation shields of platinum or gold pressed around them as developed by the National Bureau of Standards. The shielded thermocouples were first installed immediately downstream and then immediately upstream of a few unshielded thermocouples at station 2 in several typical combustors. For most of the instruments and conditions investigated, the differences in readings between the bare-wire and the pressed-shield-type thermocouples were of the order of 2 to 3 percent with the bare thermocouples reading higher when located on the upstream or flame side of the shielded thermocouples, and lower when located on the downstream side. The catalytic action of the platinum shield at temperatures below 1000° F gave higher recorded temperatures than did the gold shield, but the effect was less than 2 percent of the temperatures recorded.

Inasmuch as fluctuations in flow and temperature at any point in the turbojet-combustor outlet preclude a high degree of accuracy, it is concluded that the thermometric technique described, although probably no more accurate than 2 to 3 percent in many cases, is generally satisfactory for evaluating the altitude performance of combustors, particularly where the trend of performance with primary variables is of concern.

Experimental Procedures

Effect of inlet-air conditions on combustor performance. - For the investigation of the effect of inlet-air conditions on combustor performance, combustor-inlet pressure, temperature, and velocity were maintained constant at values typical of altitude operation of the combustor in the engine, fuel flow was varied through a wide range, and data were recorded at various fuel-air ratios. One of the three combustor-inlet parameters (pressure, temperature, or velocity) was then maintained at some new value, the other two parameters were maintained at the original values, and the fuel flow was again altered. This procedure was continued until each of the inlet-air conditions had been varied independently of the others. Combustor-inlet velocity rather than air mass flow was chosen as a parameter because the residence time of a given sample of fuel and air in the combustor is more directly related to flow velocity than to mass flow. 1393

Effect of combustor characteristics on engine performance. - In order to facilitate the determination of the effect of combustor operating characteristics on engine performance, combustors were investigated at conditions simulating operation in the design engines at various flight conditions. Simulation of the operation of a combustor in flight requires maintenance of the combustor inlet-air conditions and the combustor-temperature rise at the values encountered in the engine at the given flight condition. The required values for the inlet-air conditions and the temperature rise were obtained from engine-performance data compiled in independent wind-tunnel and flight investigations.

EFFECT OF INLET-AIR CONDITIONS ON COMBUSTOR PERFORMANCE

Effect on Combustion Efficiency and Combustor Temperature Rise

The effect of inlet-air conditions on the performance of an early U. S. annular combustor is shown in figure 9. Although the quantitative data of figure 9 apply only to the specific combustor used for the investigation, many of the general trends and phenomena that will be discussed are typical of the many gas-turbine combustors that have been investigated. The measured values of the mean temperature rise through the combustor are plotted against the fuel-air ratio, and dashed curves are included for 60, 80, and 100 percent of the theoretical temperature rise. The theoretical temperature-rise curves are based on an initial temperature of 65° F, the inlet temperature most used in the investigation. The combustion efficiency can be estimated by interpolating between the dashed curves indicating the percentage of theoretical temperature rise.

Effect of inlet pressure. - Figure 9(a) shows the effect on combustor performance of altering inlet pressure while maintaining inlet temperature and reference velocity constant. At the highest inlet pressure, the combustion efficiency (as obtained by interpolating between the dashed lines) was above 95 percent (except at fuel-air ratios below 0.007, where the fuel spray was poor) and the temperature rise through the combustor increased with increase in fuel-air ratio throughout the range investigated. Operation at fuel-air ratios higher

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than 0.018 was not attempted because local outlet temperatures exceeded the safe limits for the instrumentation. At low inlet pressures, the combustion efficiency was low and the combustor temperature rise passed through a maximum value with increase of fuel-air ratio within the range investigated. The data indicate that both the maximum temperature rise attainable and the fuel-air ratio at which it occurred decreased as the inlet pressure was decreased. As the fuel-air ratio was increased past the value giving the maximum attainable temperature rise, combustion always became unsteady and a rich-limit blow-out occurred. Changes in the slopes of the combustor operating curves were always accompanied by changes in the type of operation, as shown by the letters indicating the various types of unsteady combustion beside the data points on figure 9. Changes in the fuel-air ratio were, of course, accompanied by changes in the fuel-injection pressure, and hence in the fuel spray. The curves of figure 9 show, however, that the combustion efficiency was not markedly affected by changes in fuel-air ratio except at very high and at very low values of fuel-air ratio.

Effect of inlet temperature. - Similar data showing the effect of inlet temperature on combustor performance are presented in figure 9(b). Comparison of figures 9(a) and 9(b) shows that a decrease in the inlet temperature produced the same general effects on performance as a decrease in inlet pressure. In the intermediate fuel-air-ratio range on figure 9(b), dual performance curves were obtained and these curves are shown by dot-dash lines. This tendency of the combustor to drift into different modes of operation existed when operating at conditions giving unsteady combustion. Combustor operation followed the dot-dash curves only when the fuel flow was gradually increased from low values; sudden changes in the fuel-flow rate would cause the combustor operation to change to that indicated by the solid curves.

Effect of reference velocity. - The effect of altering reference velocity is shown in figure 9(c). An increase in the inlet velocity had the same general effects on performance as a decrease in inlet pressure or temperature.

Correlation. - The independent effects of combustor-inlet pressure, temperature, and velocity are such that a correlation results when combustion efficiency is plotted against the parameter $p_i T_i / V_r$. The quantities p_i and T_i are the static pressure and the absolute temperature at the combustor inlet, and V_r is the reference velocity computed from the air mass flow, the combustor-inlet-air density, and the maximum cross-sectional area of the combustor. Such a correlation of the experimental data obtained with one annular turbojet combustor is given in figure 10. For this particular combustor, the scatter of the data points from the curves of figure 10 is within the limits obtained in day-to-day operation at the same operating conditions. For other combustors, however, the correlations are not always so good as that shown in figure 10, but the same general curve shape is always 1393 obtained. Fuel-air ratio is known to affect combustion efficiency in some combustors; this effect is unaccounted for in the parameter. In addition, changes in combustor design, such as changes in fuel nozzle, are not in the parameter; however, the correlation is useful in predicting the efficiency of a given combustor at different operating conditions or for comparing the performance of different combustors.

Effect on Combustor Pressure Loss

The simplified analysis of combustor pressure loss presented in reference 1 indicates that the ratio of the total-pressure loss through the combustor to the effective dynamic pressure $\Delta P/q_e$ should correlate as a straight-line function of the ratio of the combustor-inlet density to the combustor-outlet density ρ_1/ρ_2. The complex geometry of most gas-turbine combustors makes accurate computation of the effective dynamic pressure impossible and a reference dynamic pressure q_r is therefore used. The straight-line correlation should be obtained using q_r if q_e bears a fixed ratio to q_r. This reference dynamic pressure is computed from the formula

$$q_r = \frac{W_a^{\,2}}{2g\rho_1 A_{max}^{\,2}}$$

where

W_a mass flow of air through combustor, (lb/sec)

g acceleration due to gravity, (ft/sec^2)

A_{max} maximum cross-sectional area of combustor housing, (sq ft)

A plot of $\Delta P/q_r$ against ρ_1/ρ_2 is shown in figure 11 for data obtained with a typical annular U. S. combustor. A straight-line correlation is obtained as predicted from theoretical considerations. The data scatter in figure 11 may be due to the fact that q_e does not have a fixed ratio to q_r.

The value for $\Delta P/q_r$ at a density ratio of approximately 1 is the value due to aerodynamic drag alone. The added pressure loss that occurs at higher density ratios is due to the momentum change of the gases flowing through the combustor.

Effect on Combustor-Outlet Temperature Distribution

The degree of nonuniformity of the combustor-outlet temperature distribution can be expressed by the parameter

$$\delta = \frac{|\Delta T_1| + |\Delta T_2| + |\Delta T_3| + \ldots\ldots\ldots |\Delta T_N|}{N}$$

where

δ mean deviation of individual thermocouple readings from average combustor-outlet temperature

ΔT_1 difference between temperature indicated by thermocouple 1 at station 2 and mean temperature at station 2

N total number of thermocouples at station 2

Values of δ obtained with a can-type combustor of British design are plotted against the combustor temperature rise in figure 12. For this combustor, δ increases with increase in combustor temperature rise throughout the range investigated. The data points on figure 12 include operation at widely different combustor-inlet-air conditions; the correlation therefore indicates that for the range investigated with this combustor the inlet-air conditions have little effect on δ. Values of the maximum deviation of a single thermocouple reading from the mean temperature are also shown in figure 12; these values follow the same trend as those for the mean deviation δ, but the curve has a greater slope. For many combustors, the values of δ, plotted as in figure 12, do not result in straight-line correlations. At conditions where unsteady or fluctuating combustion occurs the values of δ are much higher and do not follow the trends shown in figure 12. High inlet-air temperatures sometimes cause vapor lock and resultant uneven fuel distribution, which can cause δ to increase markedly.

EFFECT OF COMBUSTOR CHARACTERISTICS ON ENGINE PERFORMANCE

The relations between the basic operating variables and combustor performance can be used to explain the effect of the operating characteristics of the combustor on the performance of the turbojet engine. The altitude, the engine rotational speed, and the flight speed determine the combustor inlet-air conditions and the combustor-outlet temperature required to run the engine under steady-state conditions. As shown in the preceding section, the combustor performance in turn depends on the combustor operating conditions. 1393

Altitude Operational Limits

Flight and altitude-wind-tunnel investigations of different turbojet engines reveal that these engines have at each engine speed an altitude ceiling above which the engine cannot operate. This altitude operational limit is imposed by the combustor performance. This imposition is further demonstrated when the combustor alone is set up in a duct for study, as previously described, and is operated with inlet-air pressure, temperature, and velocity maintained at values required to simulate engine operation at particular altitudes and engine speeds. By varying the fuel flow in an attempt to obtain the temperature rise required for engine operation at each simulated altitude and engine speed, the region of altitudes and speeds where the required temperature rise can be obtained is separated from the region where required temperature rise cannot be obtained.

The combustor operating conditions for two typical turbojet engines at zero flight speed are shown in figure 13. Altitude operational limits for a typical annular combustor operating under the conditions shown in figure 13 for the engine with a compressor pressure ratio of 3 are illustrated in figure 14; good agreement is indicated between the data obtained in the altitude-wind-tunnel investigation of the complete engine and that obtained in the independent combustor investigation. The operational limitations at high engine speeds were not obtained in the combustor investigation. The combustor had fixed orifice fuel nozzles and operated on AN-F-22 gasoline. As a matter of interest, the combustor pressure along most of the operational-limit curve of figure 14 is 5 to 6 pounds per square inch absolute; the absolute limit to which combustion can be maintained was previously cited as 0.68 pound per square inch absolute for gasoline.

In order to illustrate how the influence of inlet-air conditions on the maximum temperature rise available from the combustor causes

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altitude operational limits, figures 15 and 16 are presented. Results are shown in figure 15 for which data were taken over a range of fuel-air ratios with combustor-inlet conditions simulating engine operation at seven engine speeds at an altitude of 20,000 feet. The measured values of temperature rise through the combustor are plotted against the fuel-air ratios used to obtain them. The shape of these curves is, in general, similar to those of figure 9. Curves for 100, 80, and 60 percent of theoretical temperature rise are shown in figure 15. The points where the curves surpass the temperature-rise requirement of the engine are indicated by symbols that are connected by a dotted line. At the intermediate simulated engine speeds, blow-out occurred before the temperature rise reached the engine requirement. Because a change of fuel flow in an engine is necessarily accompanied by changes in air flow and combustor-inlet pressure and temperature, these curves are not truly representative of combustor operation in an engine. The fuel flow could not be slowly adjusted in an engine to the values corresponding to the peaks of the curves without the other operating variables having changed.

The temperature-rise requirement of the engine for normal operation at various engine speeds at an altitude of 20,000 feet is shown in figure 16. Values of the maximum temperature rise attainable with various simulated engine speeds at this altitude (taken from the peaks of the curves in fig. 15) are connected by a dotted line. The resulting curve falls below the temperature-rise requirement curve between engine speeds of 38 and 68 percent rated rpm. The nonoperational range for the engine therefore exists between 38 and 68 percent rpm at an altitude of 20,000 feet and this conclusion is substantiated by figure 14.

Combustor-inlet pressures, temperature, and velocities existing in the engine at an altitude of 20,000 feet at various rotational speeds are also shown in figure 16. At engine speeds below 38 percent rated rpm, combustor-inlet pressure and temperature are low, but the inlet velocity is very low; and although the low pressure and temperature have an adverse effect on combustion, the very low velocity has a favorable effect on combustion that more than compensates for this adversity, and the combustor can deliver a temperature rise in excess of that required. As the engine is accelerated to higher engine speeds, the combustor-inlet pressure and temperature increase. At the same time, however, the velocity increases very rapidly and this change has a detrimental effect on combustion that more than counterbalances the beneficial changes in the pressure and the temperature, as is evidenced by the decrease in the temperature rise obtainable until the combustor ceases to meet the engine requirements when a speed of 30 percent rpm is reached.

As the simulated engine speed is increased through the nonoperational region, the inlet pressure and temperature increase at accelerating rates, whereas the reference velocity increases at a decelerating rate with increase in simulated engine speed. At a simulated speed of about 60 percent rated rpm, the favorable effects of inlet pressure and temperature become large enough to offset the adverse effect of the inlet velocity, and the temperature rise attainable begins to increase with simulated engine speed. At engine speeds of 68 percent rated rpm and greater, the temperature rise attainable is equal to or greater than the temperature rise required and the combustor is in the satisfactory operating range.

The effect on altitude operational limits of increasing the compressor pressure ratio without changing mass flow is shown in figure 17 for a typical turbojet combustor. This change increases the combustor-inlet pressure and temperature and reduces combustor reference velocity. Figure 13 shows the combustor operating conditions for both compressor pressure ratios, as used for figure 17.

The altitude operational limits of two turbojet engines of recent design as determined in altitude-wind-tunnel investigations of the complete engines are given in figure 18(a). The minimum speeds shown are established as the lowest speed at which an engine can be operated without encountering combustion blow-out, the minimum engine speed from which it is possible to accelerate without encountering limiting turbine temperature, or the minimum idling speed recommended by the engine manufacturer. The maximum speeds shown are the speeds at which limiting turbine-outlet temperatures are encountered. The effect of flight Mach number on operating limits is illustrated by figure 18(a). For example, in the case of engine 1, the minimum speed below "relative altitude" of about 5 and at the lowest flight Mach number was the manufacturer's recommended idling speed; however, the minimum speeds at the higher flight Mach numbers were established by combustion blow-out. Above the altitude where the abrupt change in minimum speed occurred, increases in flight Mach number had the opposite effect on minimum speed for both engines. In this region, increasing flight Mach number increased the operational limits for both engines.

Altitude-operational limits for a variety of turbojet engines as determined from combustor research are shown in figure 18(b). Because results for different engines are plotted in figure 18(b), the inlet-air conditions for one combustor are not necessarily directly comparable to the inlet-air conditions for any other combustor. The dotted portion of the curve for combustor B was estimated, inasmuch as the laboratory air facilities were unable to duplicate the combustor-inlet conditions at the simulated altitude. Figure 18(b) is intended only to show that all turbojet combustors encounter altitude operational limits. The shapes of the several curves are similar.

Temperatures Available for Acceleration

The amount by which the combustor temperature rise attainable with slow fuel-flow adjustment exceeds the temperature rise required for nonaccelerating engine operation is an approximate index of the available acceleration in that it is the maximum temperature rise available for acceleration. As previously noted, such data are not truly representative of engine performance because air flow and combustor pressure and inlet temperature also change as fuel flow is changed. The same discussion that explains the reason for altitude operational limits applies to the temperature rise available for acceleration.

In figure 19 is shown a curve where a combustor temperature rise that is 100° F above the value required for nonaccelerating engine conditions can be obtained, indicating that energy corresponding to at least a 100° F change in temperature is available for altitudes and engine speeds below this curve. Below the next curve there are 250° F or more available for acceleration, and below the lowest curve there are 500° F or more available. The data used to illustrate this performance characteristic are for the combustor and the engine having a compressor pressure ratio of 4 used for figure 17.

This performance characteristic is illustrated for a can-type combustor in figure 20. Figure 20 shows how much temperature rise in excess of that required for steady-state operation can be obtained at any simulated engine speed. The sea-level curve and the high-engine-speed portion of the curve for 30,000 feet, which are shown with no data points, are imposed by the temperature-rise limitation stated by the manufacturer for the engine used. The portions of the curves at 30,000 feet shown by data points for a slow throttle-opening time and for a 3-second opening time represent actual temperature-rise limitations imposed by the combustor. These limitations are frequently blow-out. Again, these data are simply illustrative because, with an actual engine, engine speed would not remain fixed as the throttle is opened; whereas in the combustor research for figure 20 the simulated engine speed was unchanged while establishing each temperature-rise limit. The data do show one of the difficulties encountered in trying to accelerate an engine at high altitude by rapidly increasing the fuel flow.

Specific Fuel Consumption

Combustion efficiency. - In figure 9 is shown the effect of inlet-air pressure, temperature, and velocity on combustion efficiency for ranges of fuel-air ratio. Translated into terms of altitude and engine speed, it is apparent that combustion efficiencies should be a maximum at low altitude and high engine speed where combustor-inlet pressure and temperature are a maximum and should decrease as altitude is increased or as engine speed is decreased. The variation of combustion efficiency with altitude and engine speed for the combustor of the engine with a compressor pressure ratio of 4 of figures 13 and 17 is shown in figure 21; this performance is typical of other combustors. The variation of combustion efficiency with altitude for several combustors at engine speeds of 100 and 80 percent of normal rated speed is given in figure 22. Trends, but not absolute values, of combustor-inlet-air conditions as a function of altitude are comparable among the curves in figure 22 inasmuch as each engine has an approximate compressor pressure ratio of 4 at full speed. Combustors B and C had duplex-type fuel nozzles; the other combustors had fixed-orifice hollow-spray cone fuel nozzles.

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The data show that the combustion efficiency of each of the several combustors decreases regularly and appreciably as altitude is increased. When figures 21 and 22(a) are compared with figure 22(b), combustion efficiency is seen to decrease with a decrease in engine speed. As expected, the decrease in combustor-inlet-air pressure and the decrease in combustor-inlet-air temperature as altitude is increased or engine speed is reduced cause lower combustion efficiency. The lower fuel flows that are required at high altitudes and at reduced engine speeds are achieved by reducing fuel-injection pressure. This reduction necessarily alters the characteristics of the fuel spray and thus may affect the combustion. Even the two combustors having duplex-type fuel nozzles, combustors B and C, however, display performance characteristics closely similar to the other combustors (fig. 22). The effect of fuel injection on combustion efficiency will not be discussed in detail in this paper.

Visual observations reveal that as altitude is increased at a fixed engine speed, the appearance of the combustion changes regularly from a steady yellow flame, to a white flame tinged with blue, to a blue flame tinged with white and flickering at rapid frequency, to a darker blue flame that flickers at lower frequencies and with greater amplitude, and finally to a pulsating blue flame that may suddenly and unexpectedly become extinguished. Similar observations are made as engine speed is regularly reduced at a fixed moderate or high altitude.

These regular changes in the appearance of combustion are, of course, accompanied by a regular decrease in combustion efficiency until the altitude operational limit is reached. Thus a progressive depreciation of the combustor performance occurs as altitude is increased or as engine speed is decreased. Quantitative significance is not attached to the color of the flames.

The implication of any decrease in combustion efficiency to the performance of the engine is obvious; if thrust is maintained, the specific fuel consumption of the engine will vary inversely as the combustion efficiency. Obviously, the use of higher compressor pressure ratios will aid the combustion process. A given combustor should operate at least as well at 60,000 feet with a compressor pressure ratio of 8 as it does at 45,000 feet with a compressor pressure ratio of 4.

<u>Pressure loss in combustor.</u> - As discussed under "Effect of Inlet-Air Conditions on Combustor Performance," the loss in static pressure by the gases flowing through the combustor consists of the pressure drop due to aerodynamic drag and the pressure drop required to accelerate the fluid from its inlet velocity to the outlet velocity. Although the total-pressure loss correlates best as a function of the ratio of combustor-inlet density to combustor-outlet density when expressed in terms of an inlet dynamic pressure for the combustor, it is more meaningful for engine performance to express the pressure loss in terms of the percentage of the total pressure at the combustor inlet.

Combustor pressure loss reduces the cycle efficiency of the gas-turbine power plant; in an exact analysis the combustor pressure loss is deducted from the pressure achieved by the compressor. In figure 23 are shown the effect of combustor total-pressure loss on the thrust and specific fuel consumption of two typical turbojet engines, one with a compressor pressure ratio of 6 and one with a compressor pressure ratio of 4, for a flight Mach number of 0.656 at sea-level altitude with a turbine-inlet temperature of 1500° F. The thrust and the specific fuel consumption are shown as a ratio to the thrust and fuel consumption with no combustor pressure loss. These particular curves are intended to illustrate the order of magnitude of the effect of combustor pressure loss on engine performance. A more generalized treatment is found in reference 17. Typical pressure-loss ratios are of the order of 0.04 to 0.06 for combustors of current design. A 3- to 4-percent decrease in thrust and a 3- to 5-percent increase in fuel consumption from the ideal are therefore attributed to pressure loss in the combustor. Needless to say, this pressure loss cannot be eliminated. Examination of extensive data indicates that more than one-half of the total-pressure loss is usually due to aerodynamic drag rather than momentum. The loss due to aerodynamic drag increases with any change in engine-operating conditions that increases the combustor-inlet dynamic pressure. The momentum loss increases directly with any increase of the temperature ratio across the combustor. 1393

A chart is presented in reference 18 from which the total-pressure-loss ratio together with the pressure-loss ratio due to drag and the momentum-pressure-loss ratio comprising it can be estimated for known values of the combustor-inlet weight flow, temperature, and pressure, and the combustor-outlet temperature.

Temperature Distribution

Stress considerations of the turbine blading, especially on the rotor, determine the optimum distribution of gas temperatures at the combustor outlet. Usually this optimum temperature pattern has temperatures increasing radially from turbine-blade root to tip, and temperatures uniform circumferentially. Deviations from an optimum outlet-temperature distribution either shorten the turbine blade life or necessitate operating the gas-turbine engine at low temperature and thus at low power, or at low power and at low efficiency. The effect of combustor design on temperature profile will not be discussed in detail in this paper. Ordinarily the combustor inlet-air conditions of pressure, temperature, and velocity do not affect the outlet-temperature distribution unless these conditions are made so unfavorable for combustion that unsteady burning is encountered. It is shown in the section "Effect of Inlet-Air Conditions on Combustor Performance" that the nonuniformity of the combustor-outlet temperature distribution may be expected to increase as combustor temperature rise is increased. The data in table I show that the effects of altitude and engine speed on outlet-temperature distribution for two annular combustors are about as expected; in general, the mean temperature deviation from the average combustor-outlet temperature increases with increase in engine rotor speed and increase in altitude. Mean temperature deviation increases at engine speeds requiring high combustor temperature rise and is greatest at the high-engine-speed, high-altitude condition where not only a high combustor temperature rise exists, but combustor performance is depreciated to low efficiency and to unsteady burning because of the high-altitude condition.

Effect of Water in Inlet Air

Investigations were conducted on a single can-type combustor to determine the maximum quantity of water that could be injected into or upstream of the combustor without reducing the attainable combustor-outlet temperature below the value required for engine operation. At each altitude investigated, water was sprayed into the inlet-air duct at a point 62 inches upstream of the combustor until the combustor-outlet temperature required for full-speed engine operation without water injection at that altitude could not be achieved regardless of fuel flow. These data are shown in figure 24 as the ratios to air mass flow of water flow, fuel flow, and total liquid flow at the limiting liquid-flow conditions. Air flow, pressure, and temperature at the combustor inlet were maintained at the same values during the water-injection investigations as specified for combustor operation without water addition. Also shown in figure 24 are the points at sea level where water added at the inlet of a full-scale engine with the same combustors was noted as interfering with the combustion process and causing afterburning through the turbine.

At least below altitudes of 5000 or 10,000 feet, even the very heaviest rains recorded at ground stations do not add sufficient water content to the combustor-inlet air to impair the combustion process.

CONCLUDING REMARKS

Several general statements can be made about the relations between the operating variables and the steady-state performance of combustors for aircraft gas turbines. These statements are based on the accumulated results of systematic research with a number of different combustors wherein factors such as inlet-air pressure, temperature, and velocity and fuel-air ratio are independently varied and the effect on performance characteristics noted. The primary difficulty in such research is that it is virtually impossible to vary operating parameters independently without affecting one or more significant factors in the combustor; for example, in varying fuel-air ratio, either air flow or fuel-injection pressure must necessarily also be varied. At present these parameters have not been generalized to a combustor equation in a totally satisfactory manner although research of this type is in progress. Nevertheless, the general trends observed should prove useful for further research and development and provide insight into the effect of combustion on turbine-engine performance. Research on problems of design, fuels, and the transient phenomena of ignition and acceleration have helped to develop the concepts and understanding of the combustion processes in the turbine engine, but these investigations are outside the scope of the present paper.

CONCLUSIONS

1. The following conclusions pertain to aircraft gas-turbine combustors of the liquid-fuel type in general use at present and operating with the fuels for which they were designed - gasoline or kerosene: 1393

(a) The combustion efficiency and the maximum obtainable temperature rise decrease if: (1) the combustor-inlet pressure is decreased, (2) the combustor-inlet temperature is decreased, or (3) the combustor reference velocity is increased. A useful correlation usually results when the combustion efficiency is plotted against the product of inlet pressure and temperature divided by a reference velocity.

(b) As predicted from theoretical considerations, the ratio of the total-pressure drop through the combustor to a reference dynamic pressure correlates as a straight-line function of the ratio of the inlet-air density to the outlet-gas density with pressure drop increasing as the density ratio increases.

(c) The average deviation from the mean combustor-outlet temperature is unaffected by inlet-air conditions except in certain combustors where these conditions cause pronounced unsteady burning or partial vaporization of the fuel within the injection system.

2. At a given flight speed, engine speed, and altitude, the combination of combustor inlet-air pressure, temperature, and velocity, and the outlet temperature required determine the combustor performance. Thus, from the observations described, the effect of combustor performance on engine performance is noted as follows:

(a) For any aircraft turbine engine, at each engine speed an altitude exists above which the engine will not operate because of a limit in temperature rise available from the combustor.

(b) The same factors that impose an altitude operational limit impose a limit on the temperature rise available for acceleration at altitudes near the operational limit.

(c) Combustion efficiency decreases with increasing altitude and decreasing engine speed.

(d) Examination of extensive data on combustors of current design indicates that pressure loss in the combustor to the extent of 4 to 6 percent of the combustor-inlet total pressure is usual, reducing the thrust of a typical turbojet engine by 3 to 4 percent and increasing the fuel consumption by 3 to 5 percent for full-speed operation at

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sea level. More than one-half of the pressure loss is usually caused by aerodynamic drag in the combustor.

(e) The average deviation from the mean combustor-outlet temperature is generally the greatest at high-altitude, full-engine-speed conditions where high outlet temperatures and depreciated combustion are encountered.

(f) Even the heaviest rainfall recorded at ground stations does not add enough water to the inlet air to impair the performance of the combustor.

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TABLE I

MEAN TEMPERATURE DEVIATION AT VARIOUS FLIGHT CONDITIONS

Altitude (ft)	Engine speed (percent rated rpm)	Mean temperature deviation at combustor outlet, °F	
		Combustor 1	Combustor 2
35,000	33	128	---
	42	118	188
	67	---	162
	92	165	---
	100	200	---
40,000	33	---	215
	42	---	197
	50	182	---
	67	205	158
	92	201	---
	100	202	---
45,000	33	---	224
	42	---	196
	50	153	199
	58	115	188
	67	211	164
	83	230	---
	92	240	255
	100	248	---
50,000	67	225	180
	75	247	---
	83	245	---
	92	281	293
	100	275	---
55,000	67	---	195
	75	171	---
	92	304	345
	100	355	---
60,000	75	---	255
	83	---	308
	92	---	335
	100	425	---

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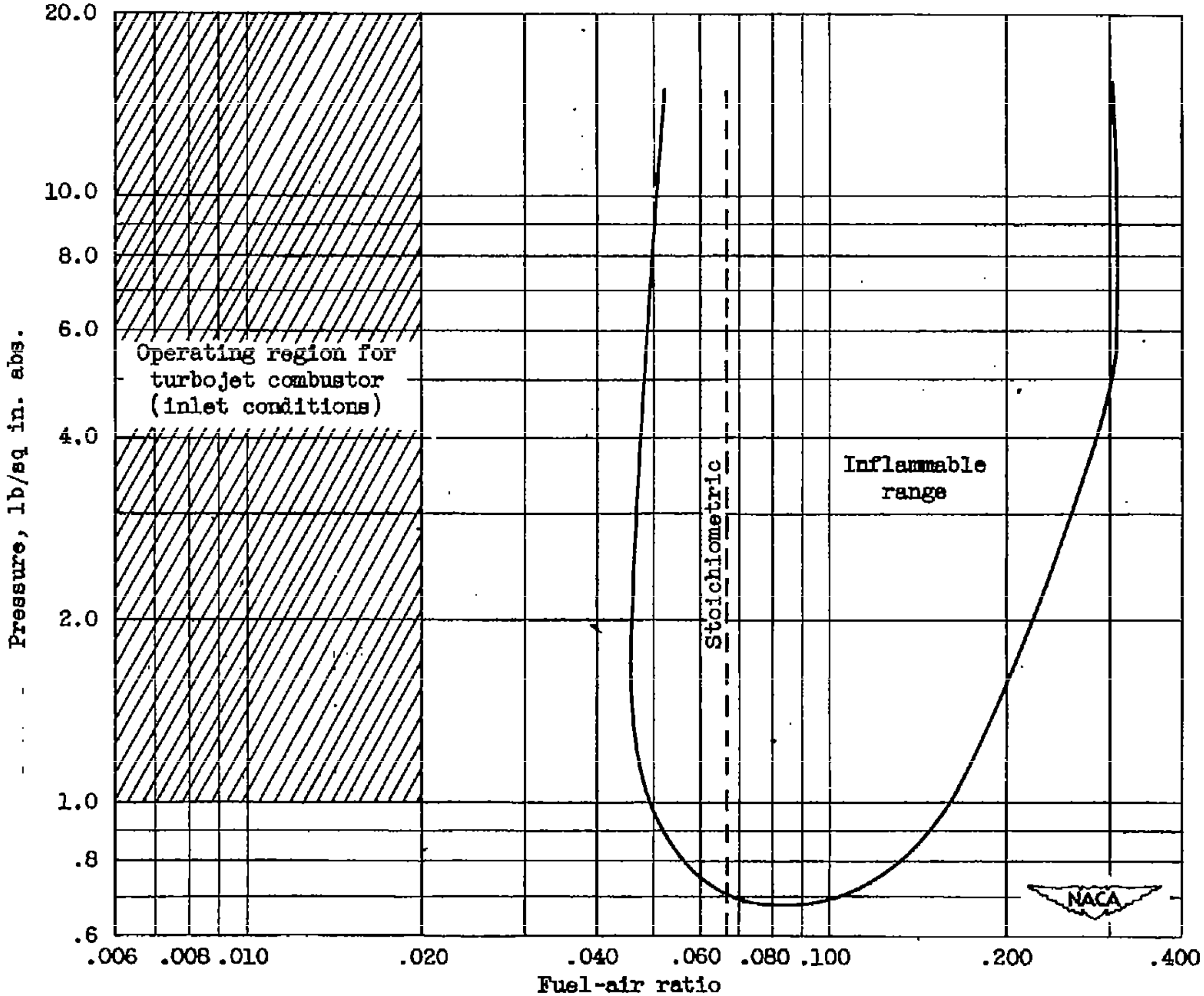

Figure 1. - Inflammability limits of 100-octane gasoline in air compared with approximate operating conditions required by typical turbojet combustor. Data from reference 9.

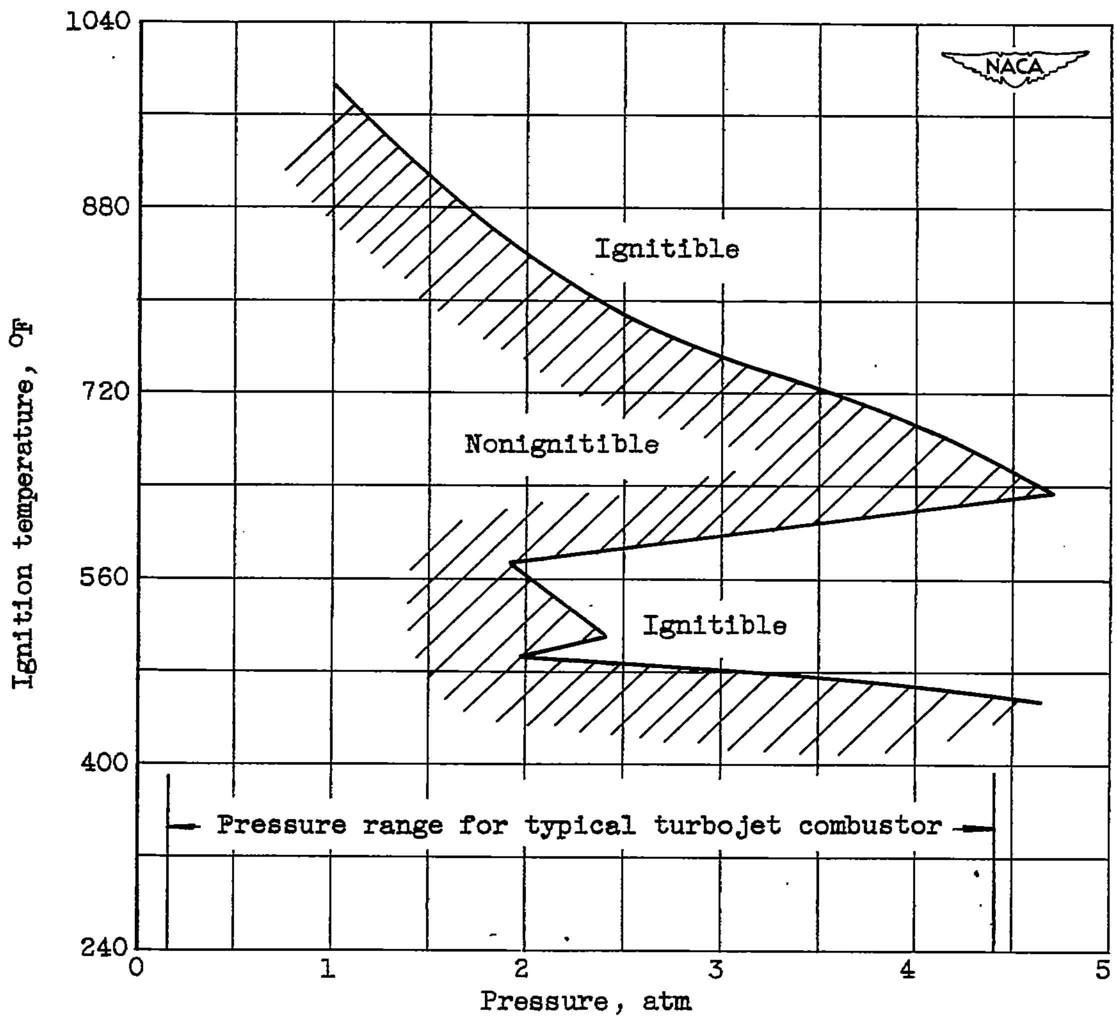

Figure 2. - Spontaneous ignition temperature as function of pressure for 1.8-percent hexane in air (stoichiometric, 2.2 percent). Data from reference 10.

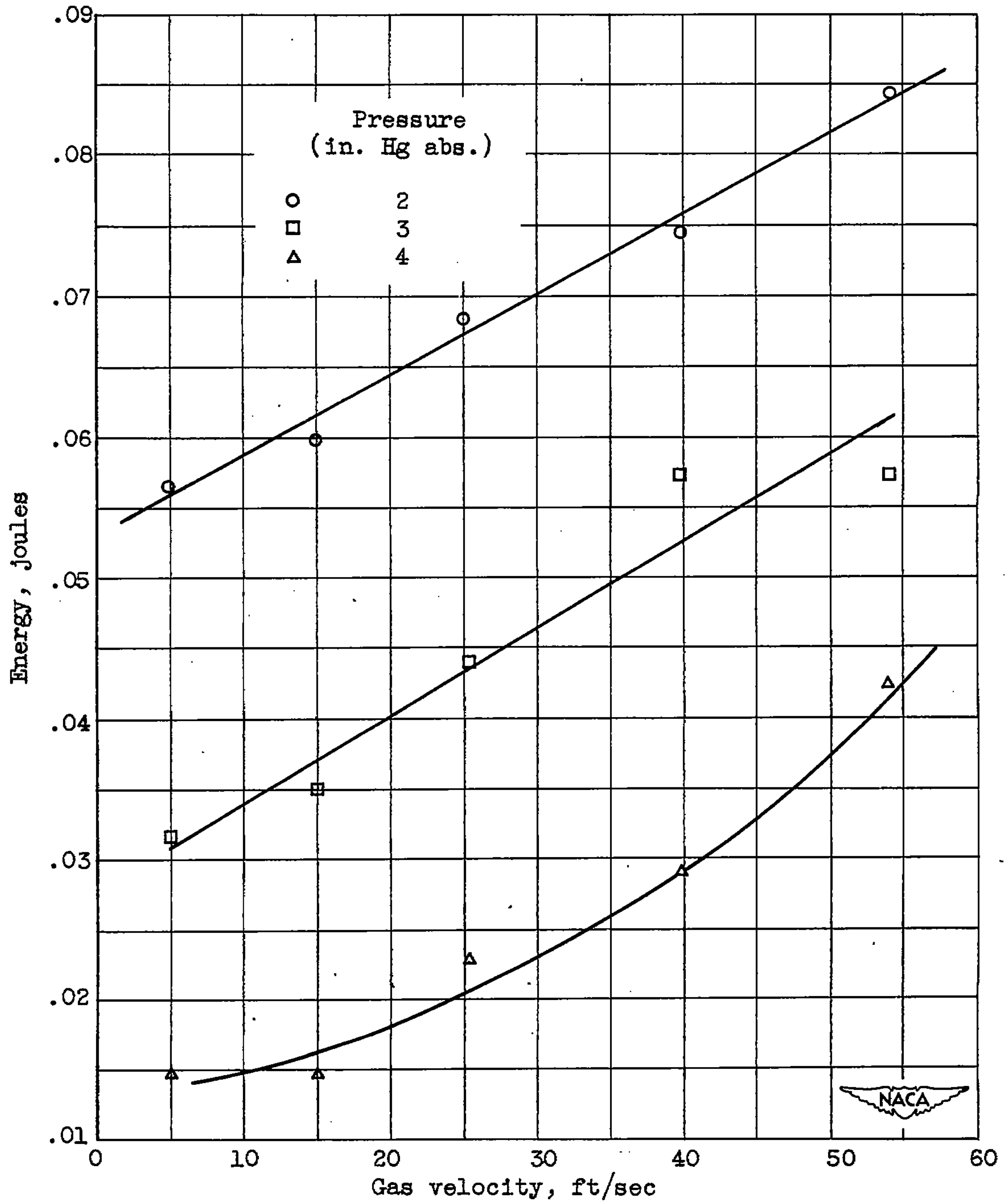

Figure 3. - Effect of gas velocity and pressure on energy required for ignition of propane-air mixture. Fuel-air ratio, 0.0835; spark duration, approximately 600 to 800 microseconds; electrode dimensions: diameter, 3/16 inch; gap, 1/4 inch. Data from reference 11.

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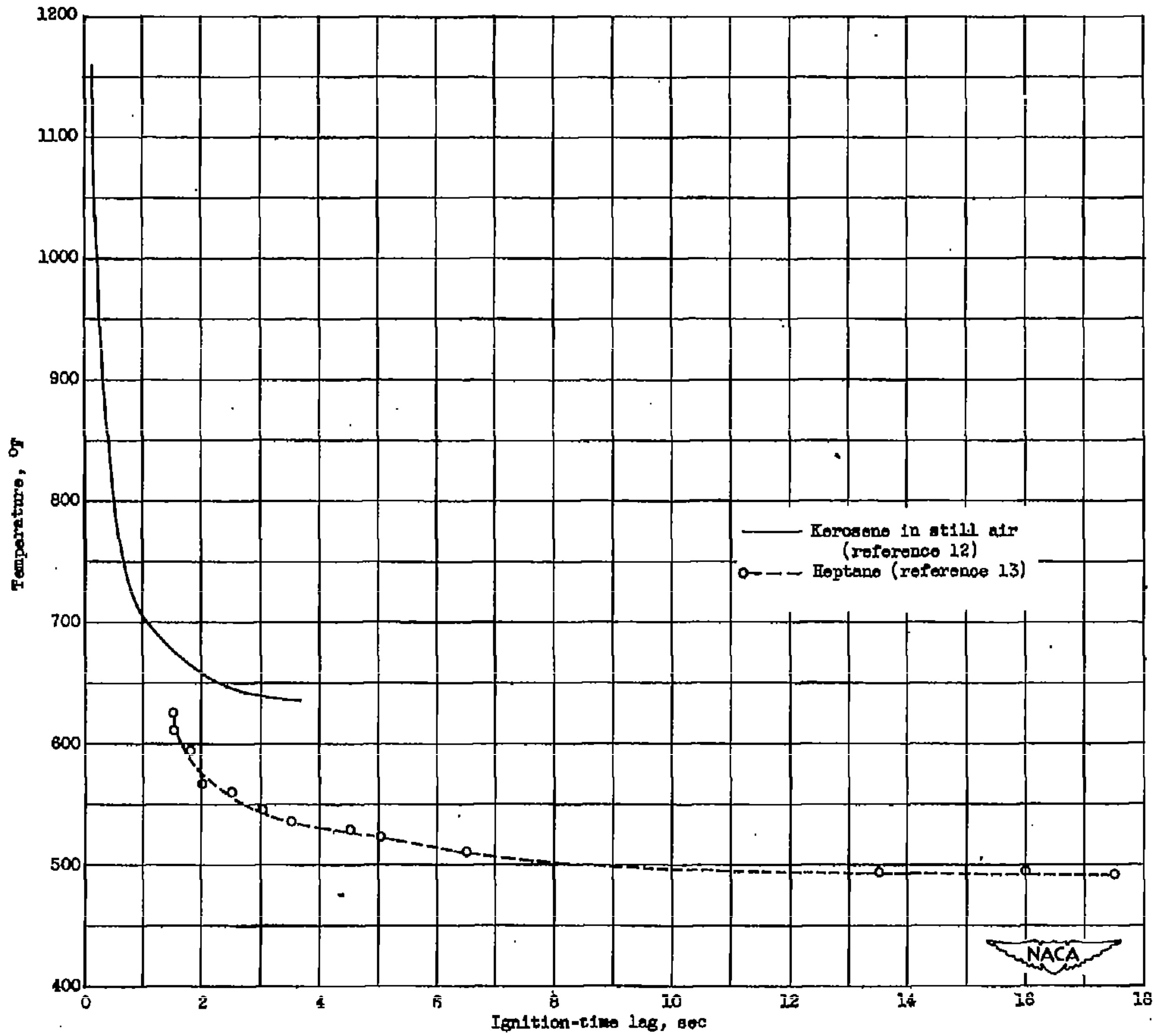

Figure 4. - Ignition delays of droplets (approximately 0.1-in. diam.) of kerosene and heptane in air at pressure of 1 atmosphere.

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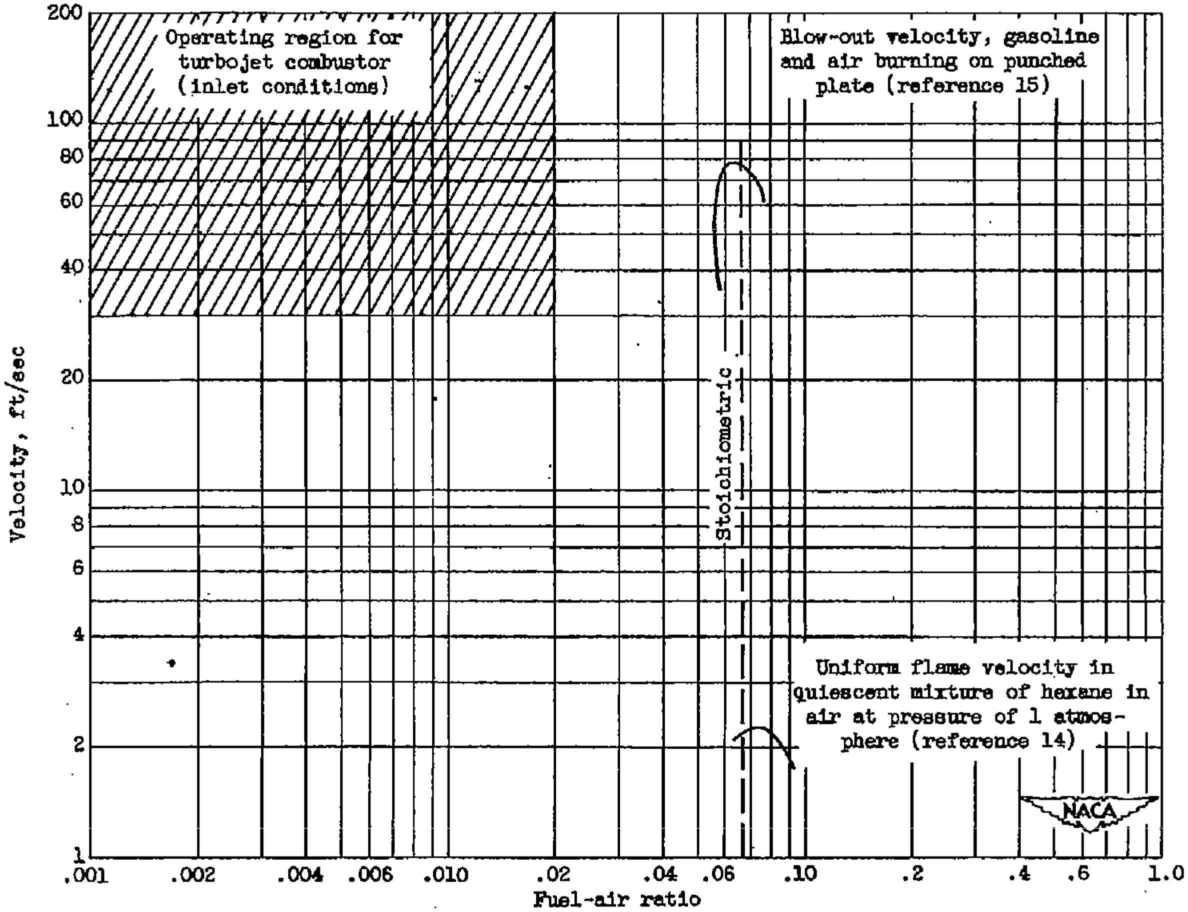

Figure 5. - Uniform flame velocity and blow-out velocity for homogeneous combustible mixtures compared with approximate conditions for operation required by typical turbojet combustor.

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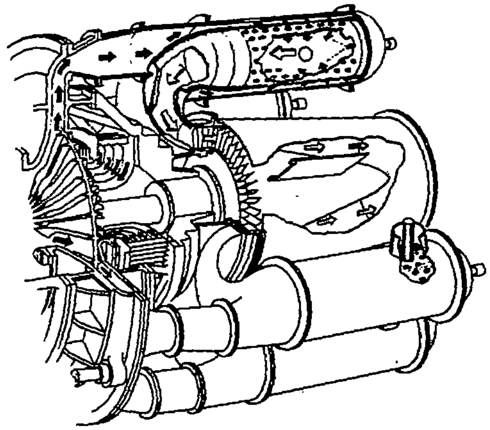
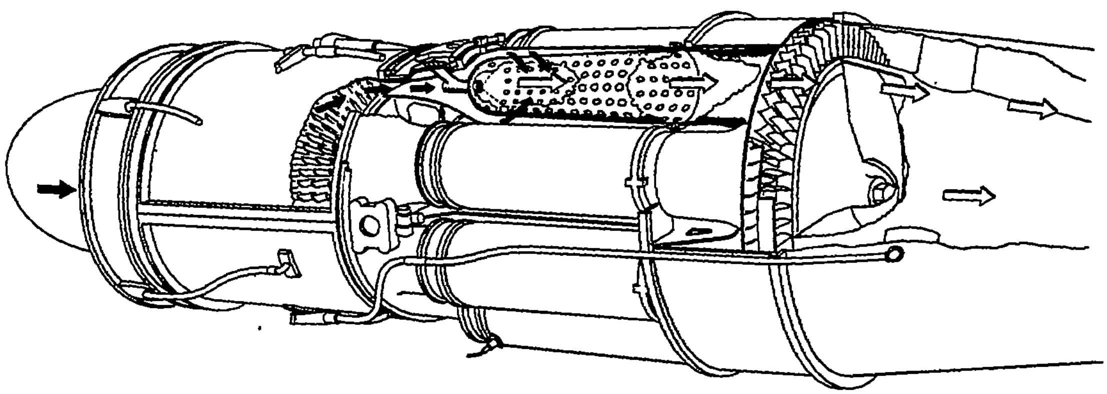


(a) Reverse-flow can-type.



(b) Straight-through can-type.

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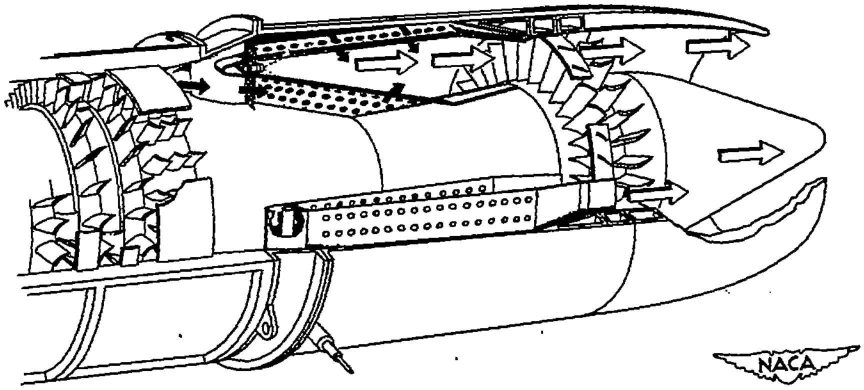

(c) Annular-type.

Figure 6. - Typical arrangements of combustors for aircraft gas-turbine engines.

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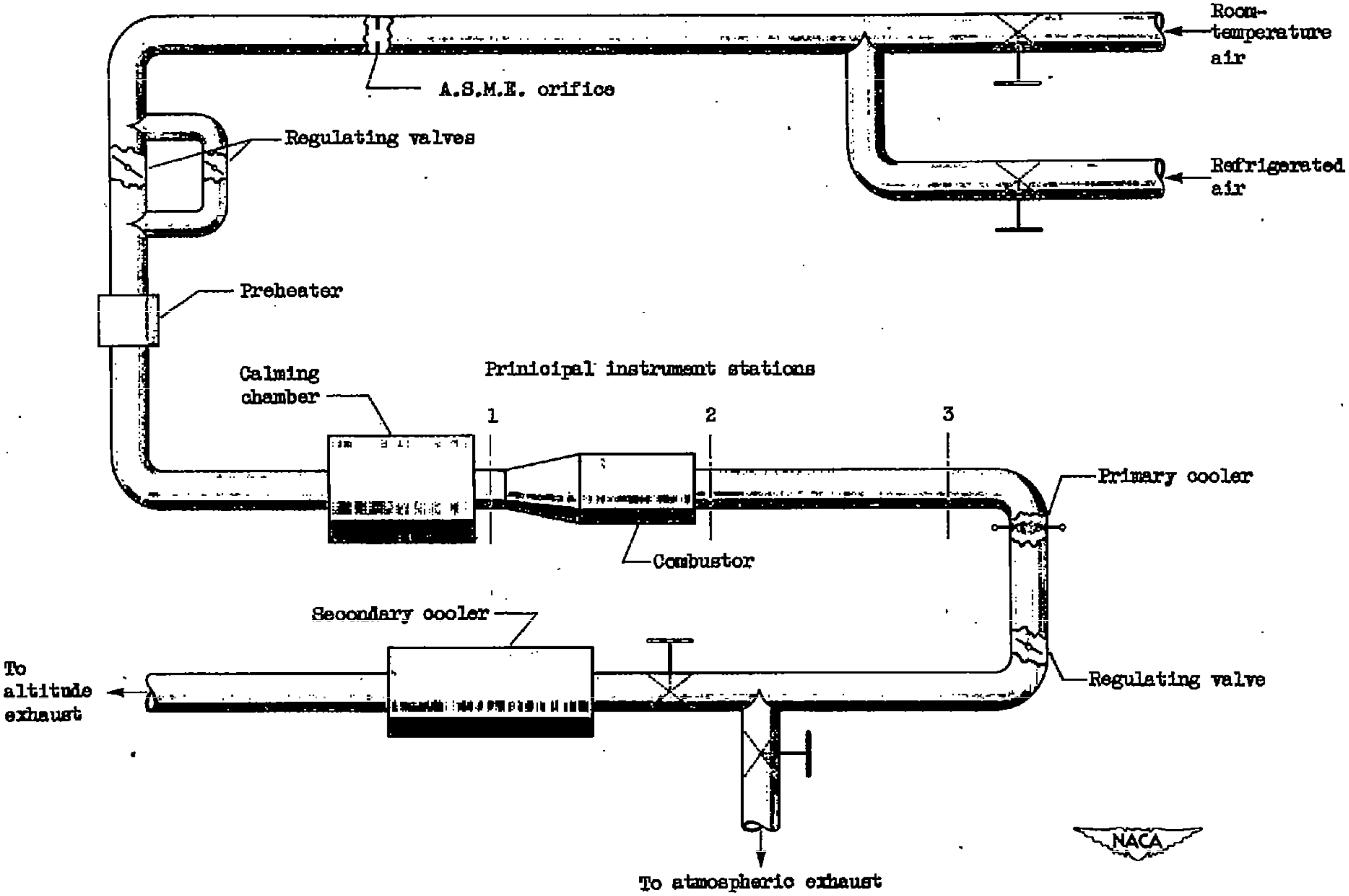
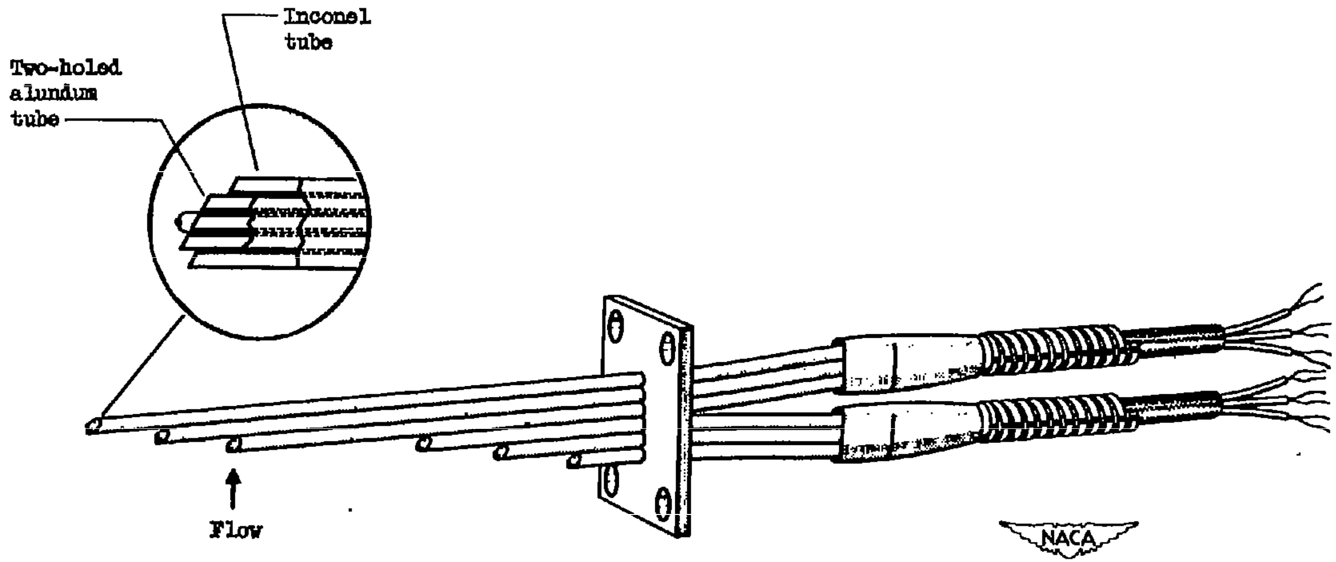
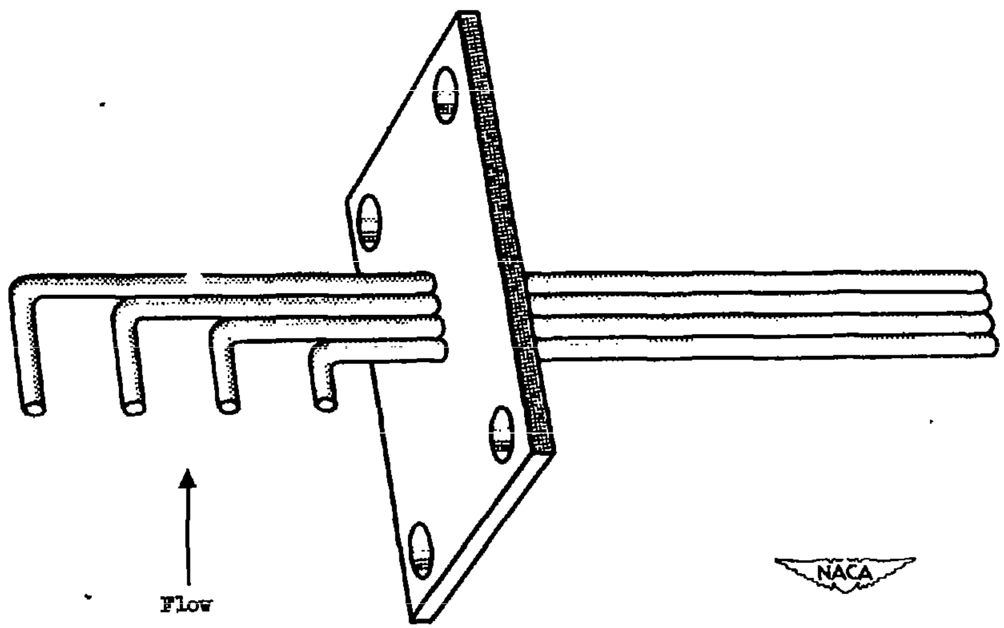


Figure 7. - Diagram of typical apparatus for investigation of gas-turbine combustor performance.



(a) Six-point thermocouple rake.

Figure 8. - Typical instrumentation used in gas-turbine-combustor investigations.



(b) Four-point total-pressure rake.

Figure 8. - Continued. Typical instrumentation used in gas-turbine-combustor investigations.

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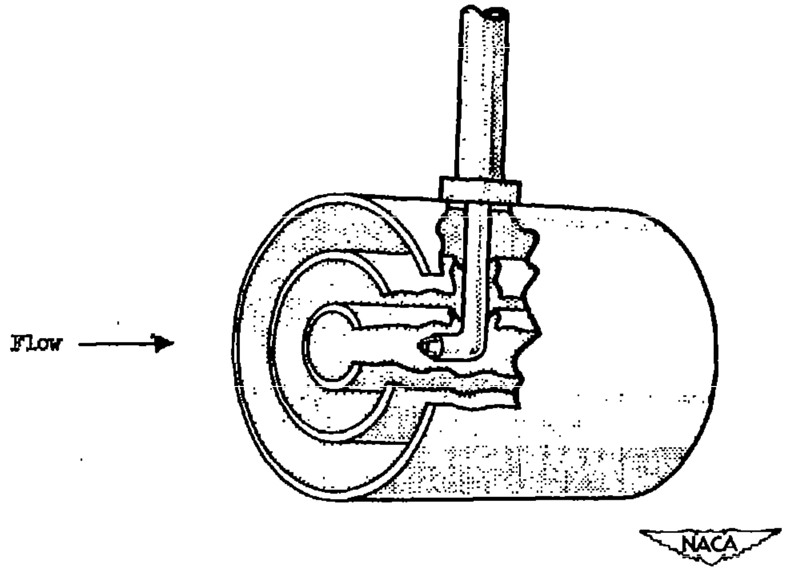

(c) Multiple-shielded thermocouple.

Figure 8. - Concluded. Typical instrumentation used in gas-turbine-combustor investigations.

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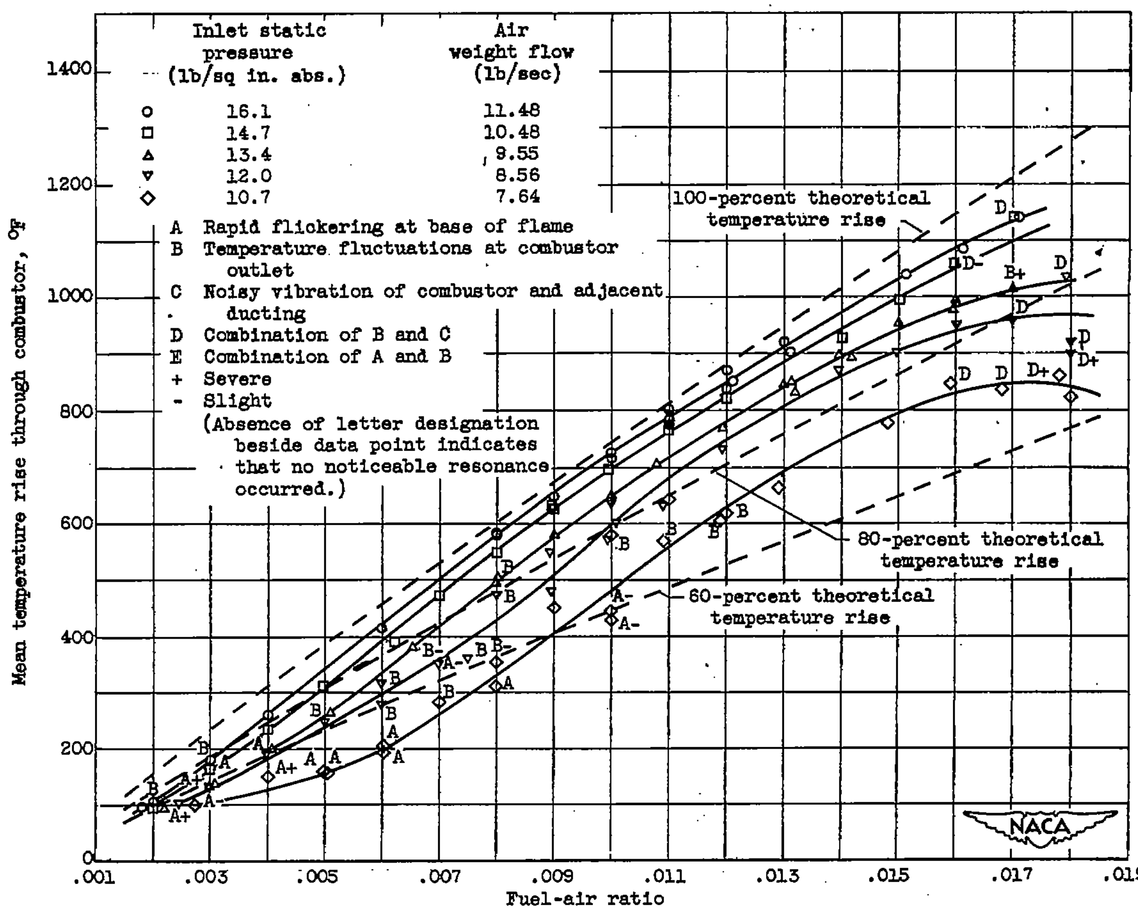

(a) Effect of altering combustor-inlet static pressure. Inlet temperature, 65° F; reference velocity, 85.1 feet per second.

Figure 9. - Variation of mean temperature rise through combustor with fuel-air ratio for combustor-inlet conditions independently altered from values typical of those encountered in altitude operation of a gas-turbine engine.

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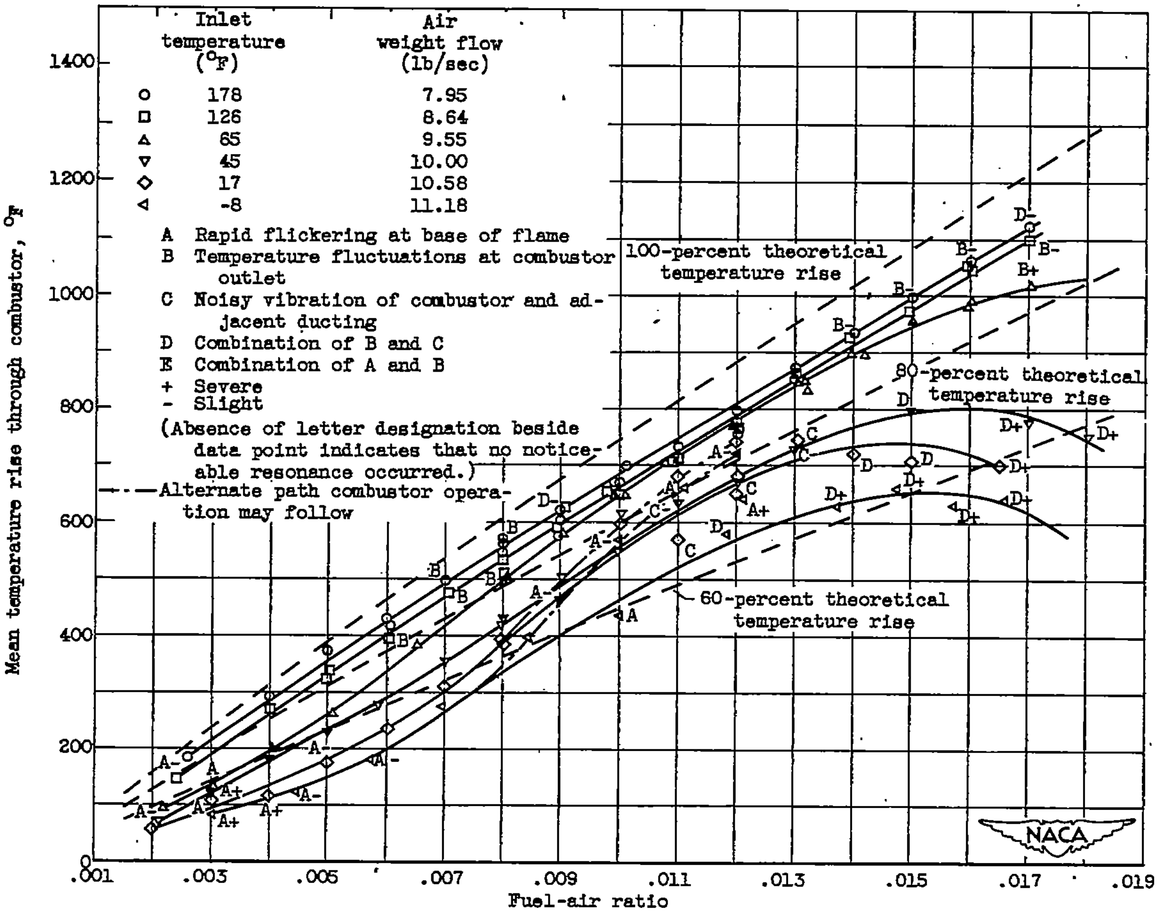
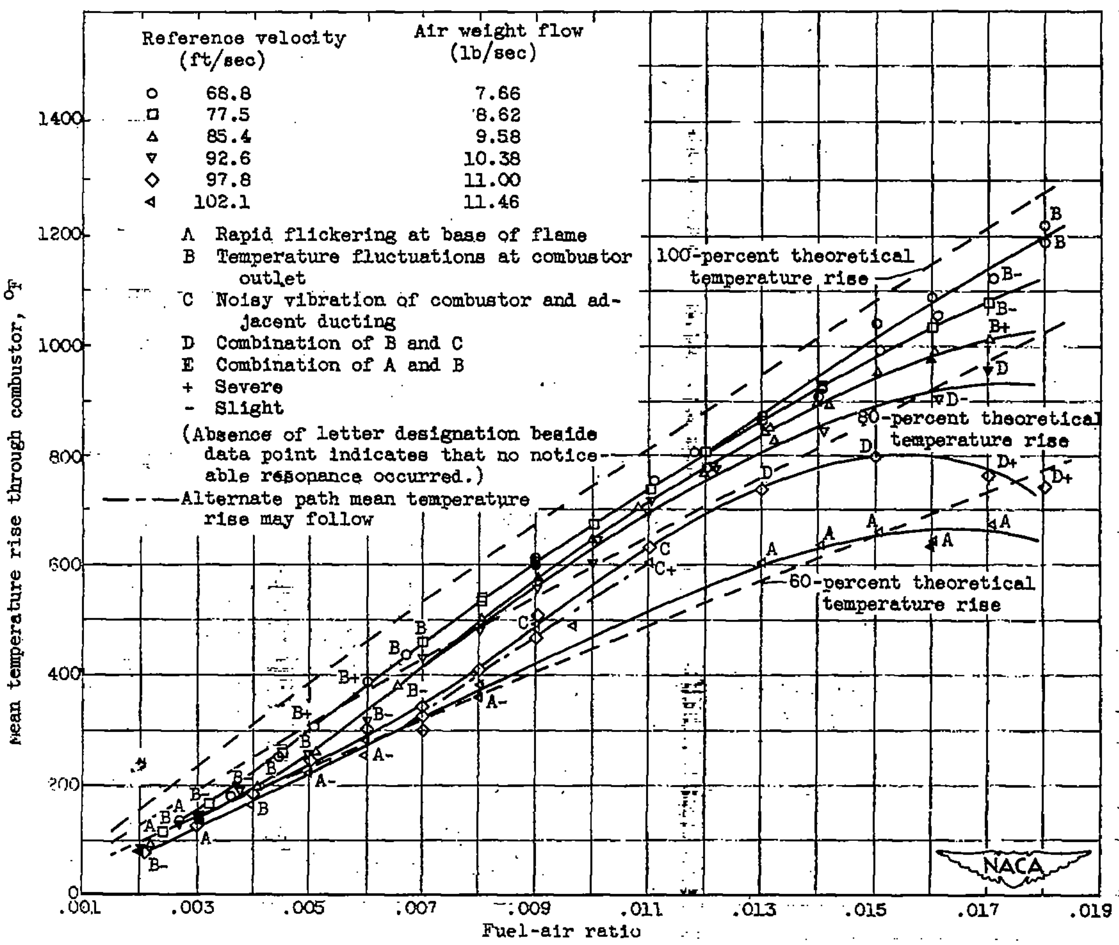


(b) Effect of altering combustor-inlet temperature. Inlet static pressure, 13.4 pounds per square inch absolute; reference velocity, 85.4 feet per second.

Figure 9. - Continued. Variation of mean temperature rise through combustor with fuel-air ratio for combustor-inlet conditions independently altered from values typical of those encountered in altitude operation of a gas-turbine engine.



(c) Effect of altering combustor reference velocity. Inlet temperature, 68° F; inlet static pressure, 13.4 pounds per square inch absolute.

Figure 9. - Concluded. Variation of mean temperature rise through combustor with fuel-air ratio for combustor-inlet conditions independently altered from values typical of those encountered in altitude operation of a gas-turbine engine.

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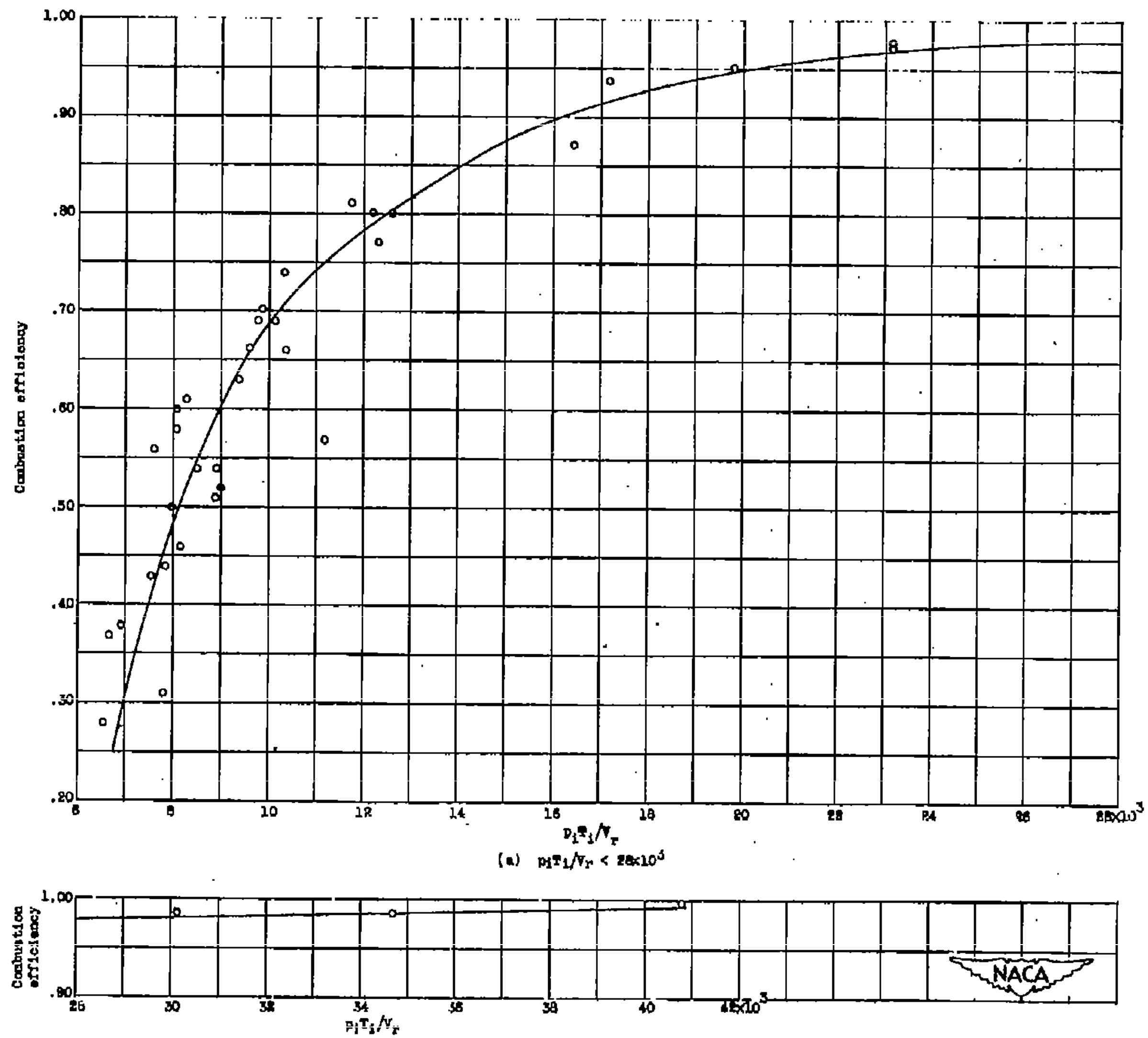

(a) $p_iT_i/V_r < 28\times10^3$

(b) $p_iT_i/V_r > 28\times10^3$.

Figure 10. - Correlation of experimental data.

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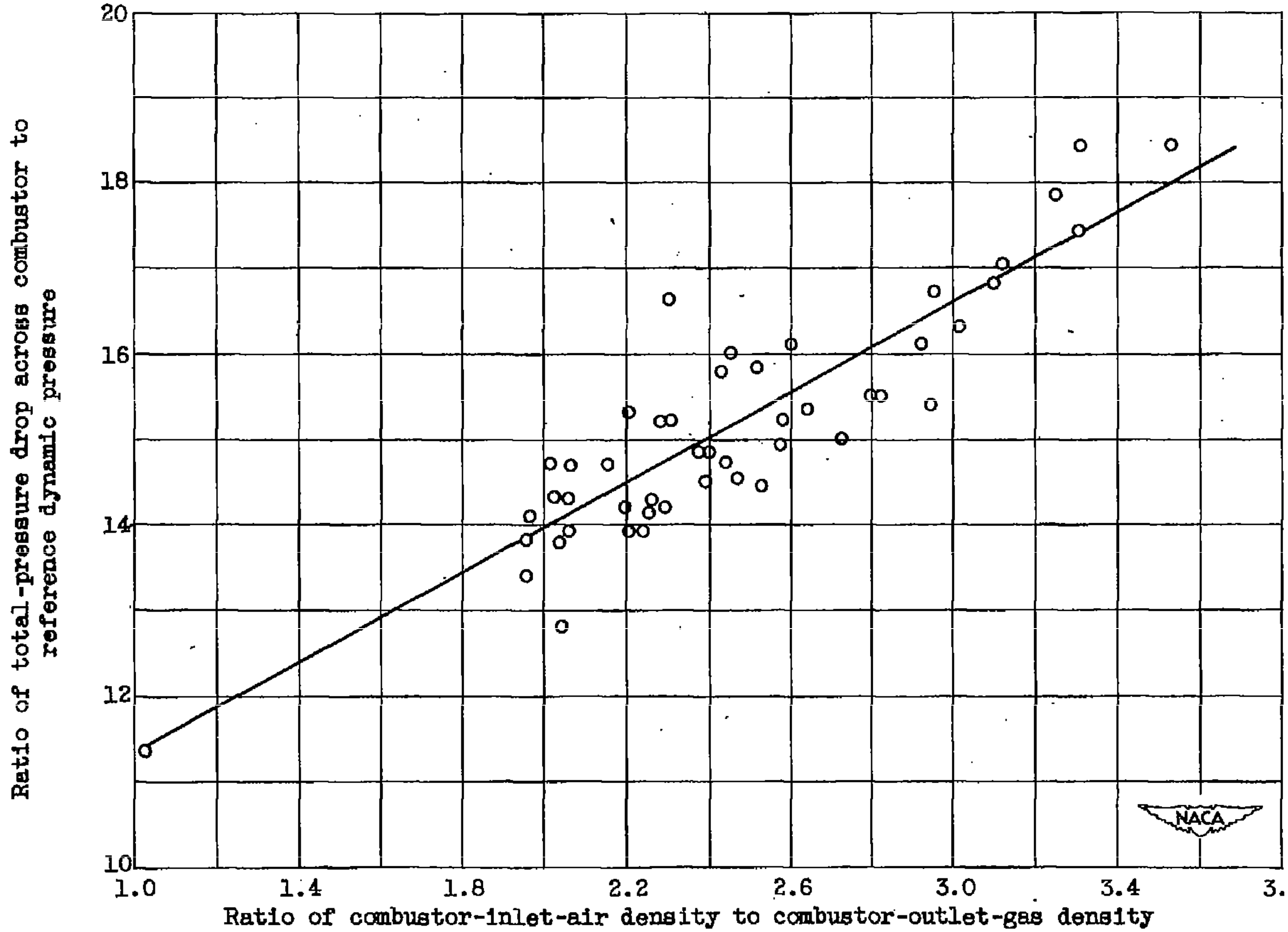

Figure 11. - Pressure losses in gas-turbine combustor.

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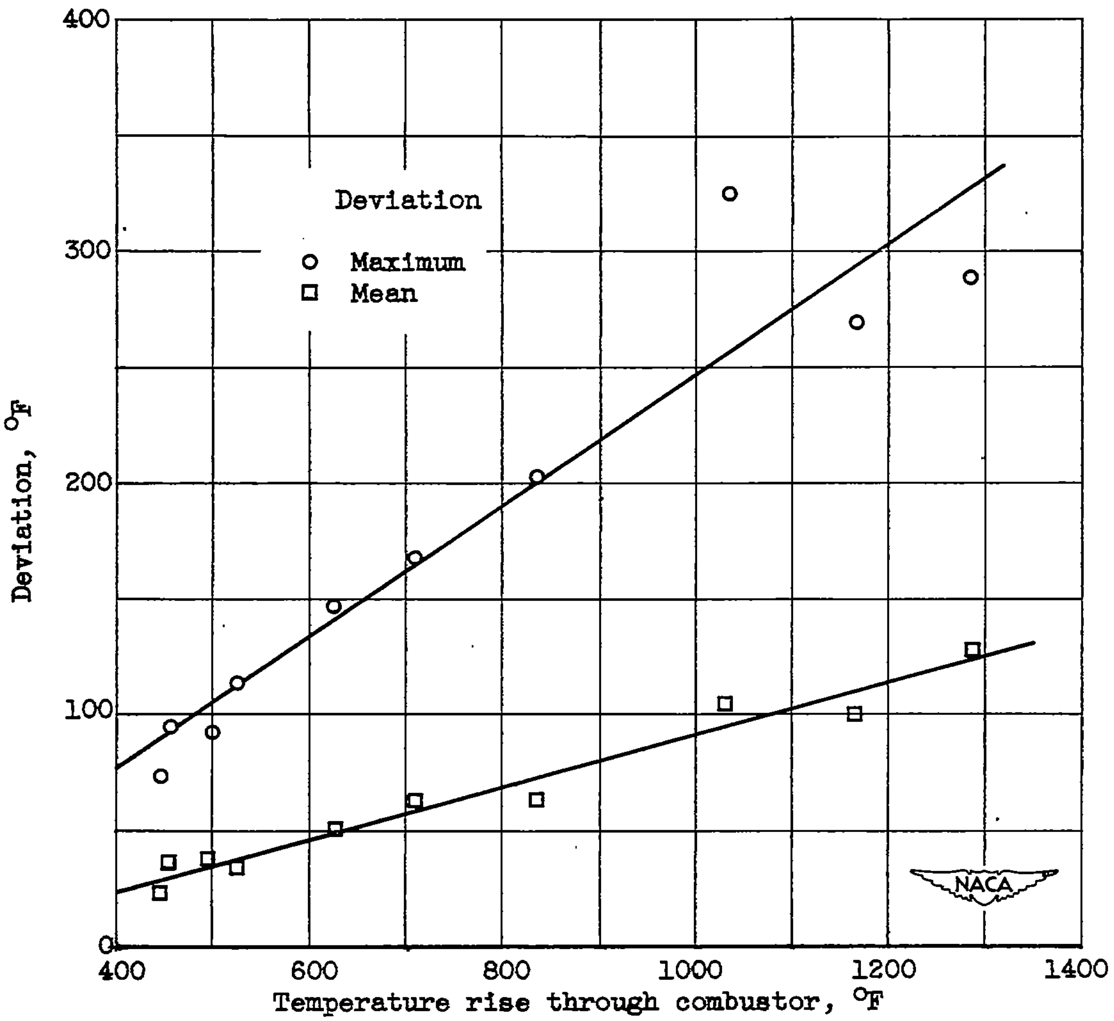

Figure 12. - Combustor-outlet-temperature uniformity expressed by deviations of point readings of outlet temperature from the mean combustor-outlet temperature.

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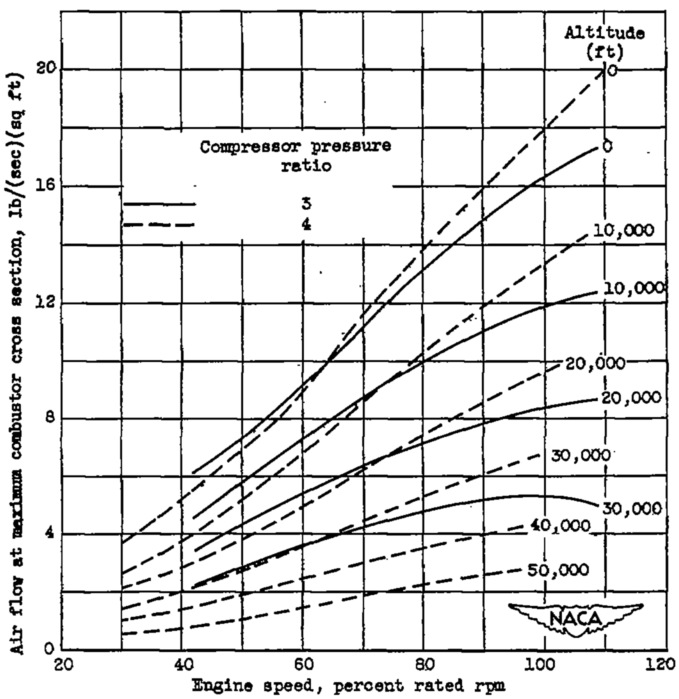

(a) Air flow.

Figure 13. - Inlet-air conditions and required outlet temperatures for combustor of two typical turbojet engines. Zero flight speed.

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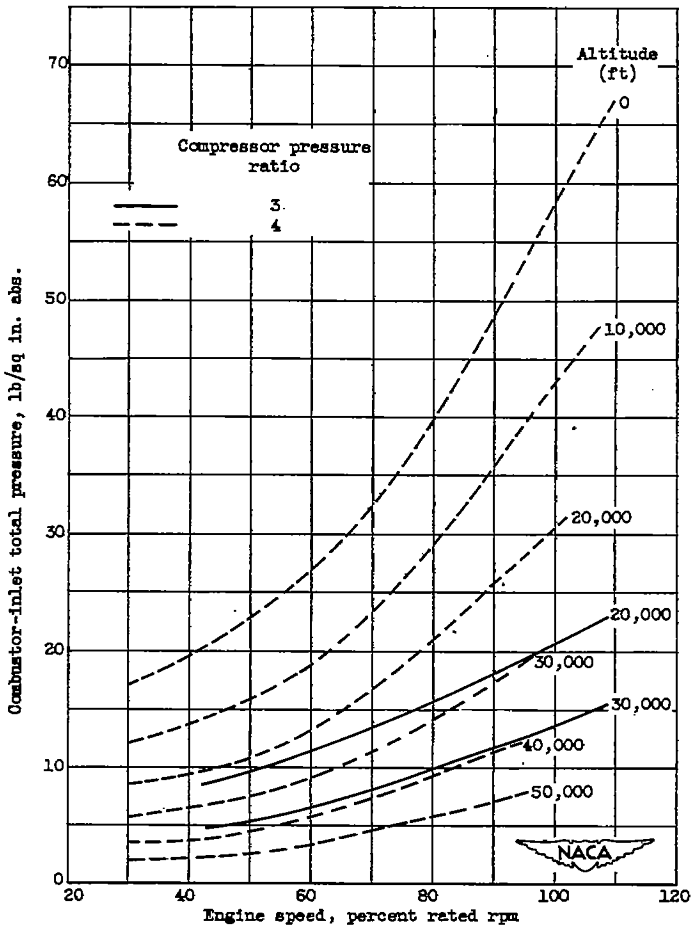
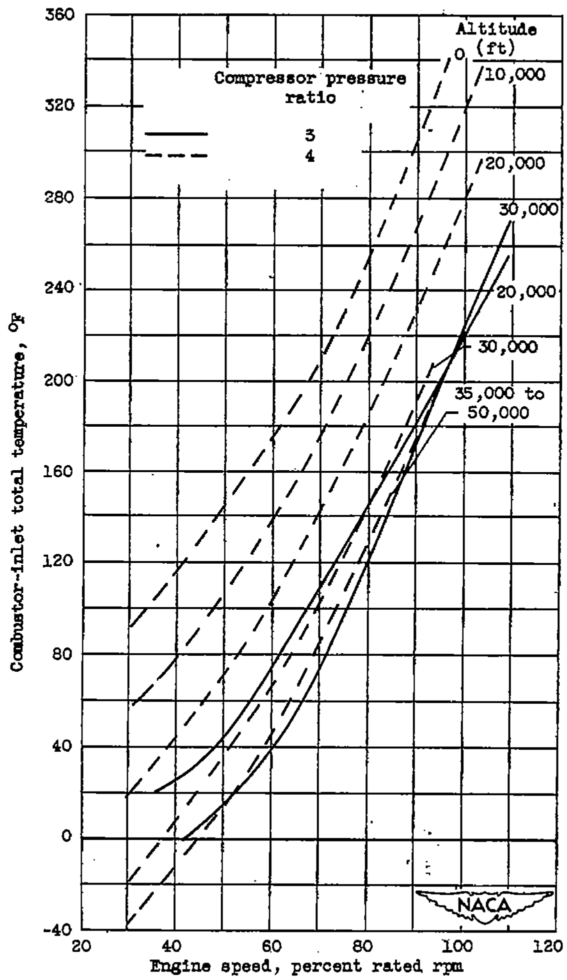


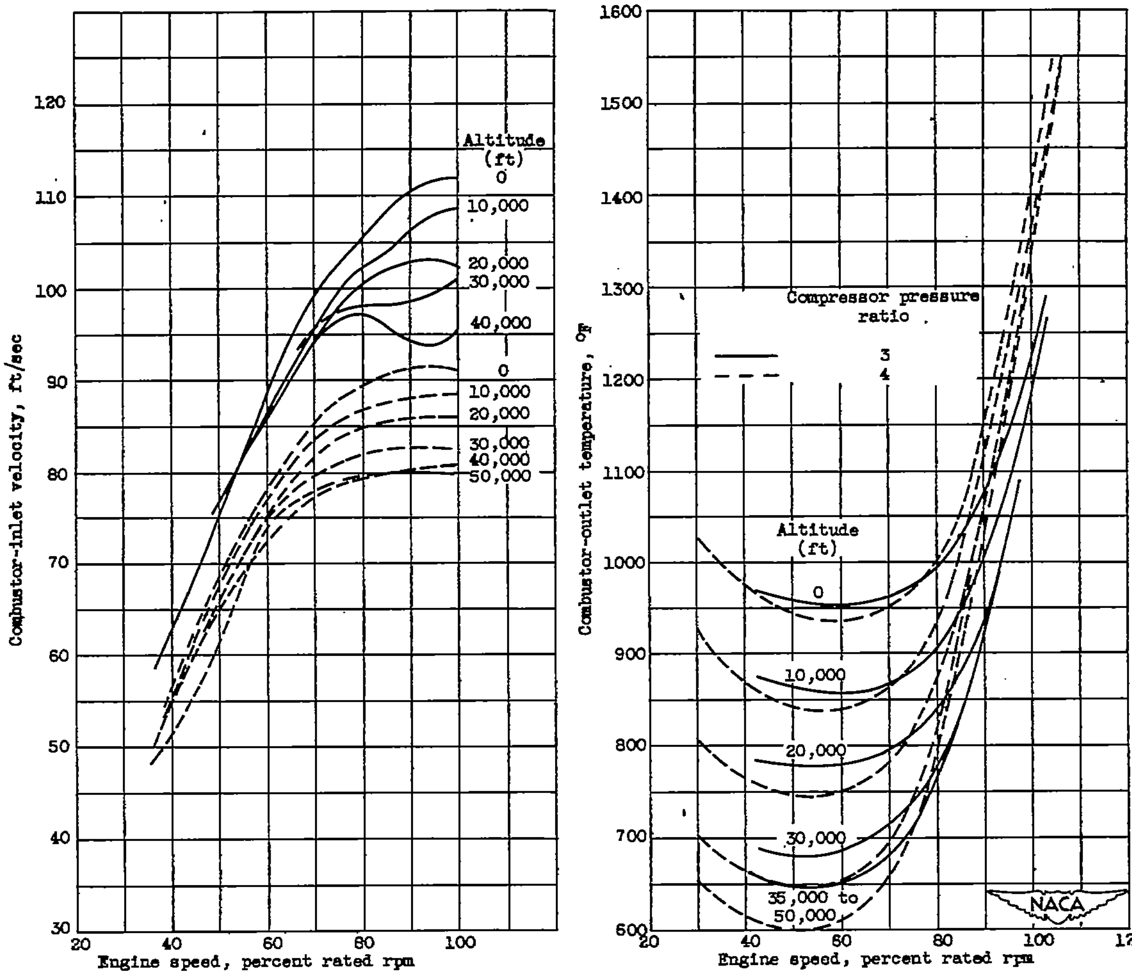
(b) Inlet-air pressure.

Figure 13. - Continued. Inlet-air conditions and required outlet temperatures for combustor of two typical turbojet engines. Zero flight speed.



(c) Inlet-air temperature.

Figure 13. - Continued. Inlet-air conditions and required outlet temperatures for combustor of two typical turbojet engines. Zero flight speed.



(d) Combustor-inlet velocity.

(e) Outlet-gas temperature.

Figure 13. - Concluded. Inlet-air conditions and required outlet temperatures for combustor of two typical turbojet engines. Zero flight speed.

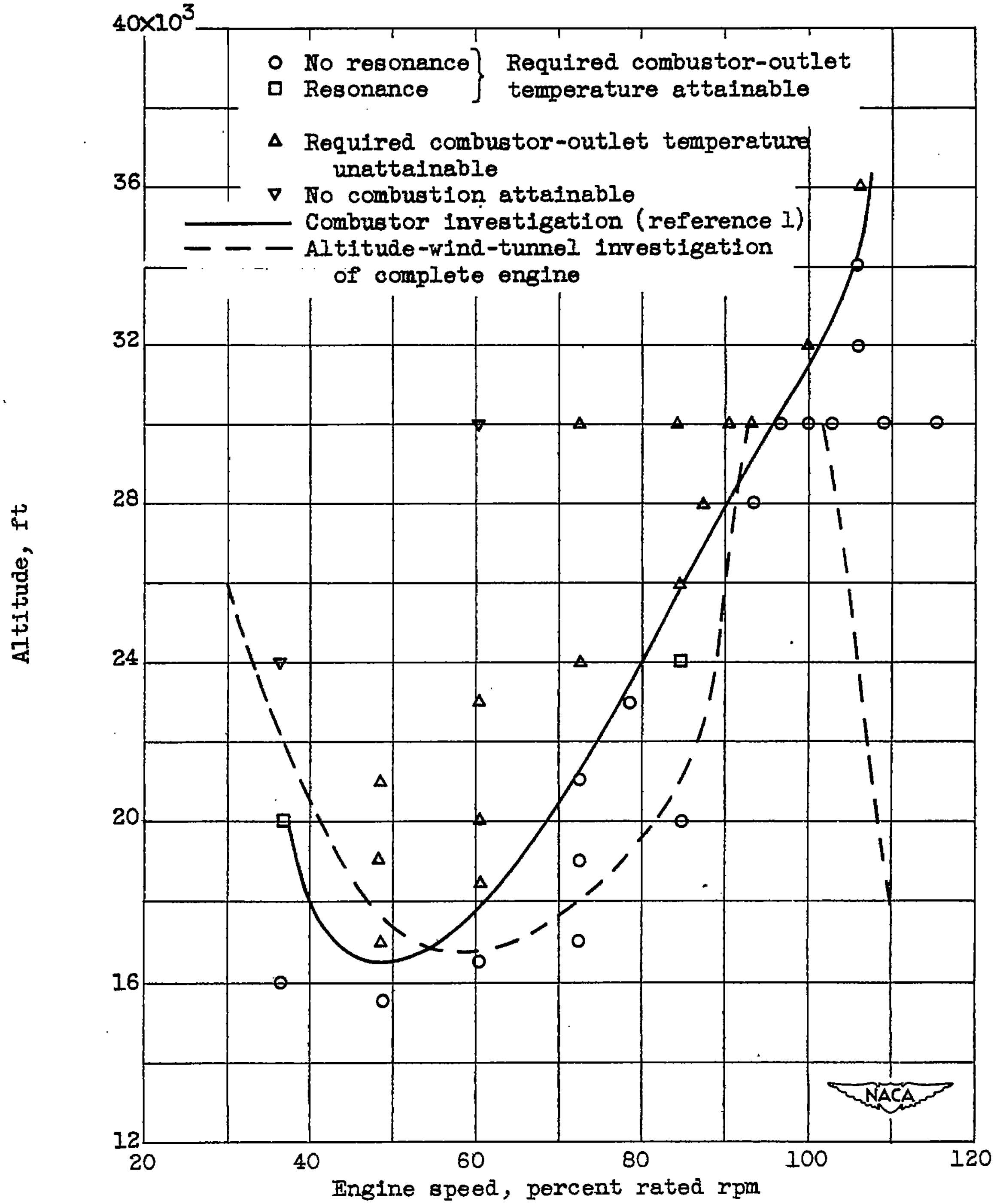


Figure 14. - Altitude operational limits of combustor under conditions simulating static operation of engine with compressor pressure ratio of 3. Fuel, AN-F-22.

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[redacted]

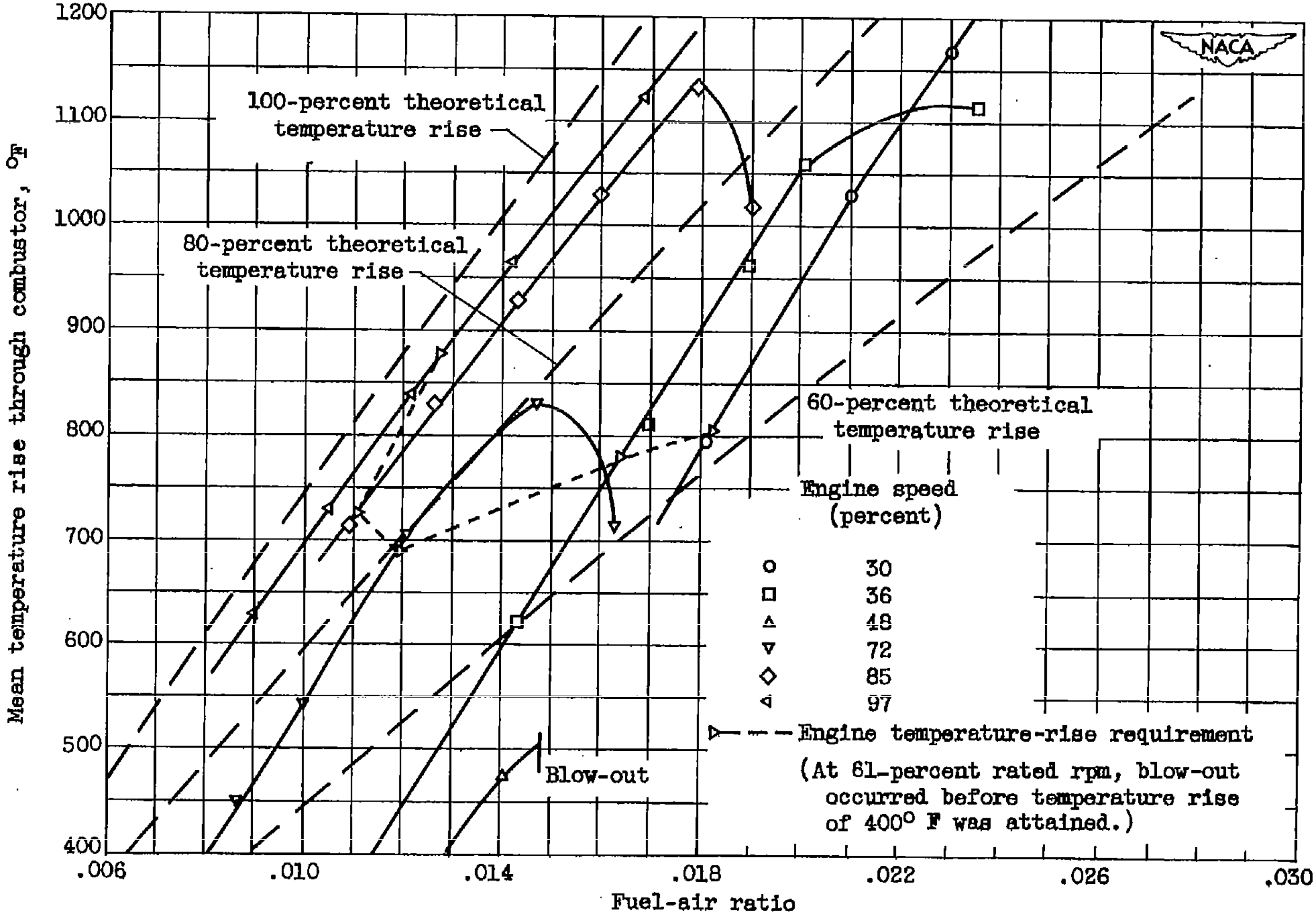

Figure 15. - Operation of combustor with inlet conditions corresponding to various engine speeds. Altitude, 20,000 feet; compressor pressure ratio, 3; flight speed, 0; fuel, AN-F-22.

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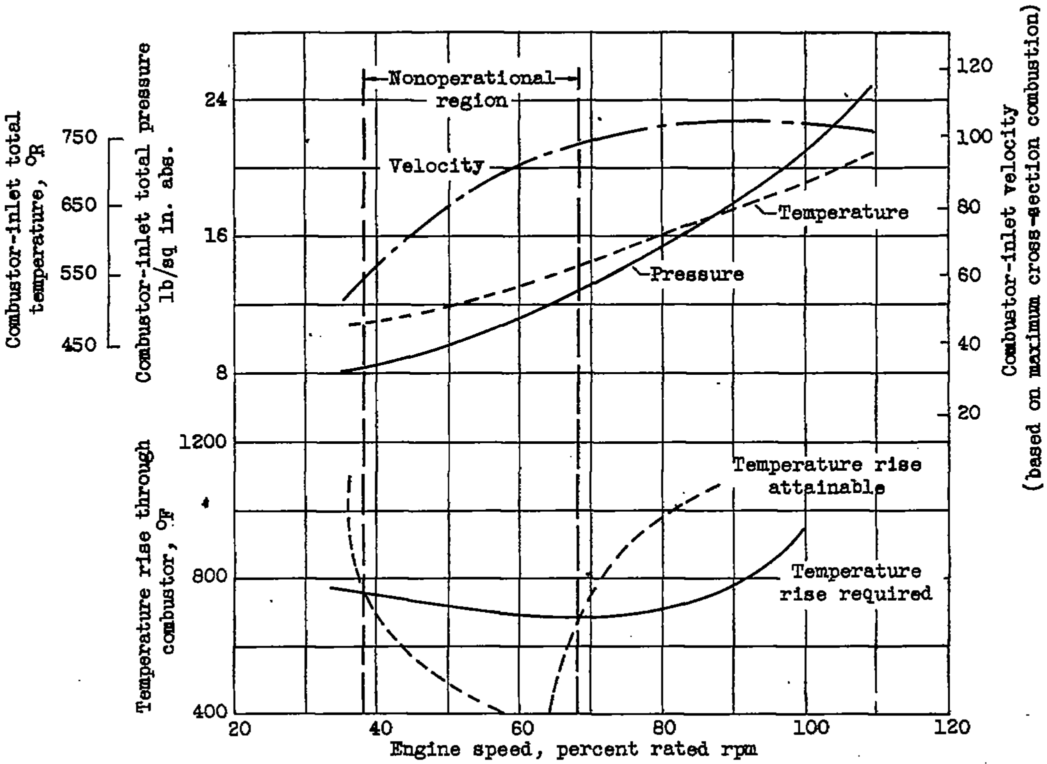

Figure 16. - Combustor operating conditions in turbojet engine at various engine speeds. Altitude, 20,000 feet; compressor pressure ratio, 3; flight speed, 0; fuel, AN-F-22.

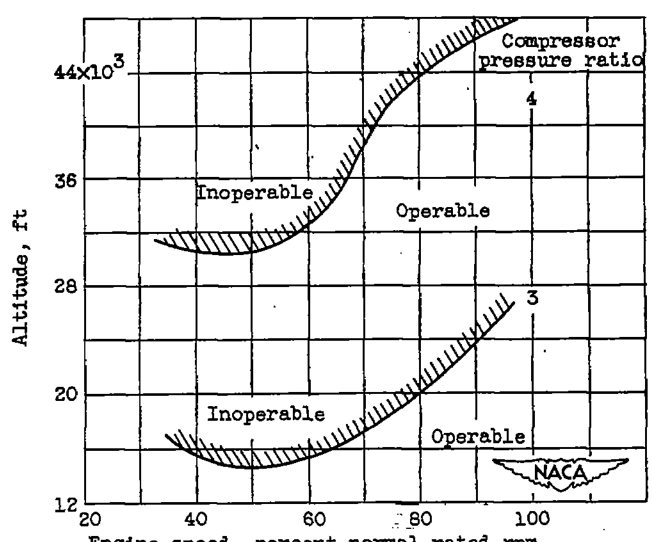

Figure 17. - Effect of compressor pressure ratio on altitude operational limits for typical turbojet combustor. Fuel, AN-F-28.

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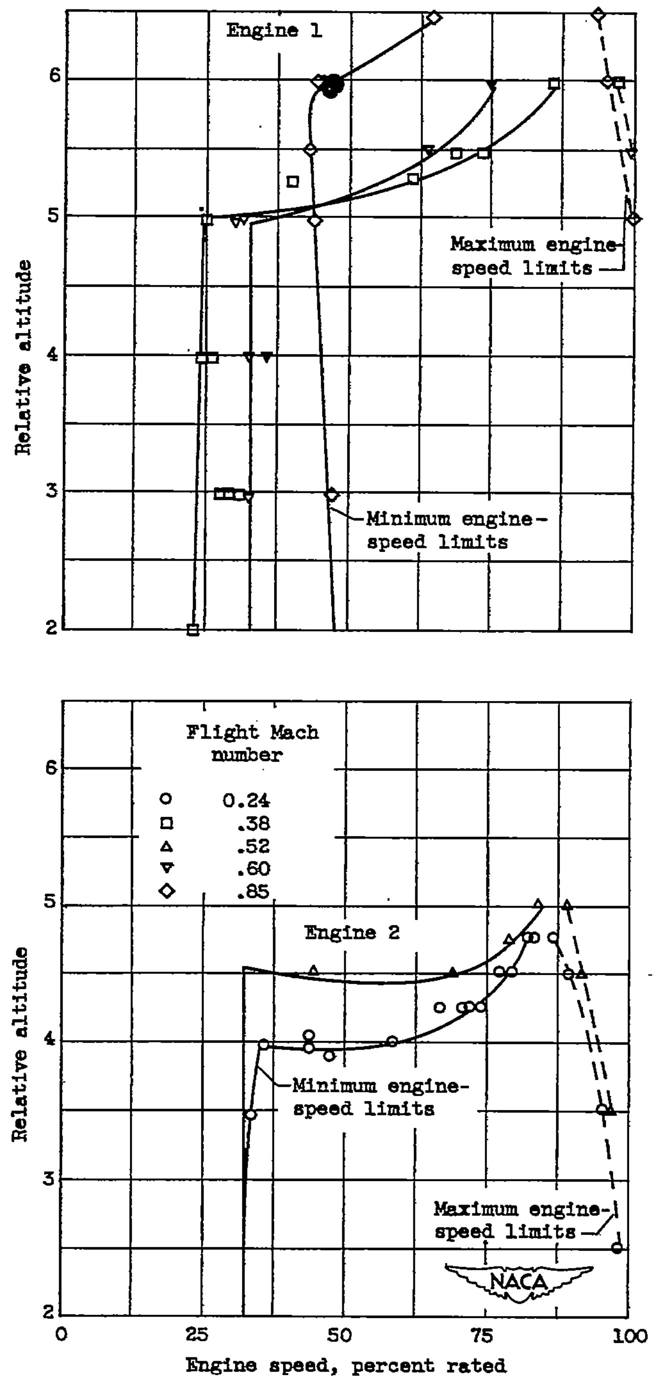
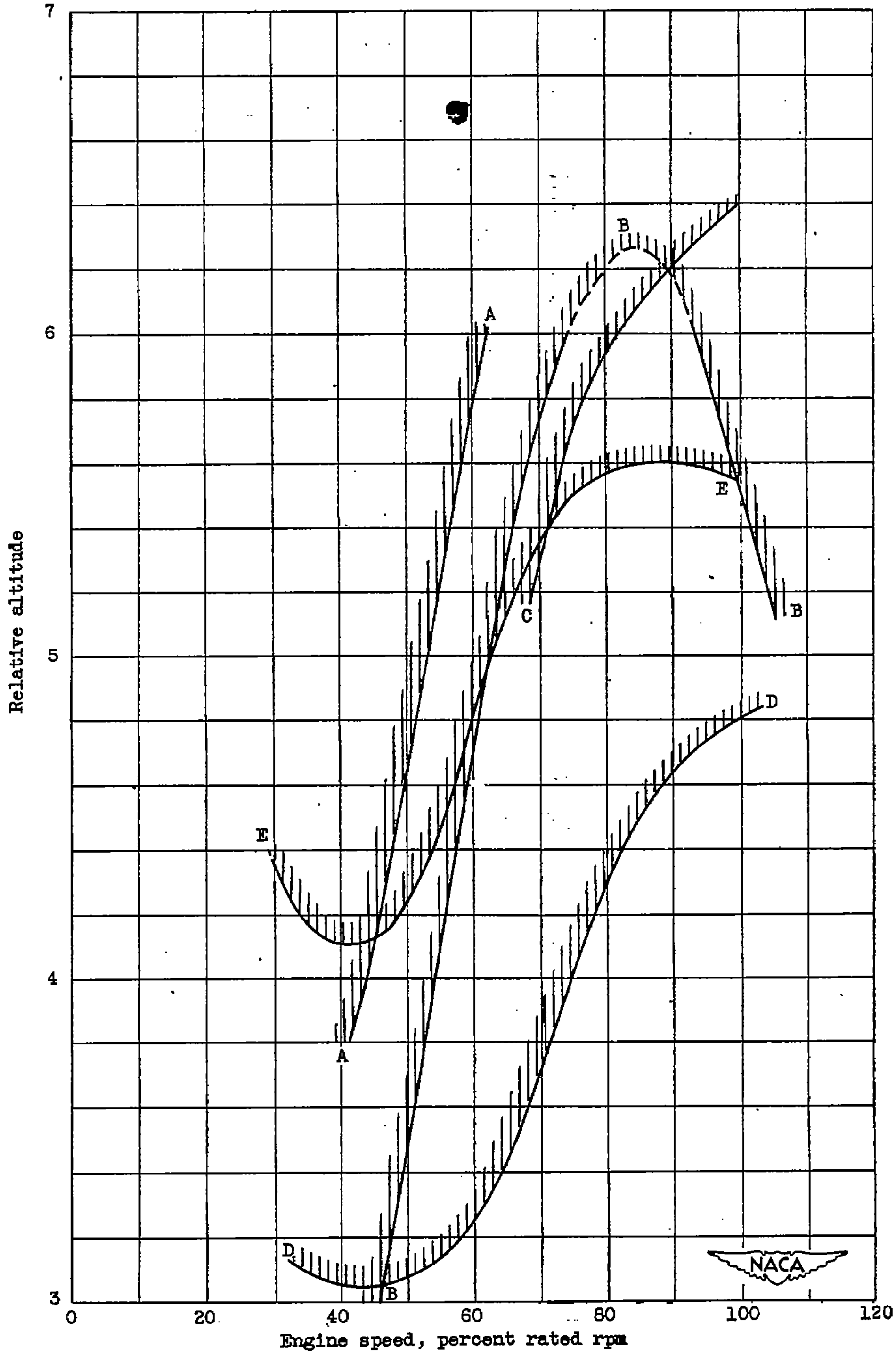


(a) Effect of flight Mach number for two typical turbojet engines as determined from altitude-wind-tunnel tests.

Figure 18. - Altitude operational limits.



(b) Limits for several turbojet engines as determined from combustor research at simulated conditions for zero flight speed. Hatching indicates side of curve on which engine is inoperable.

Figure 18. - Concluded. Altitude operational limits.

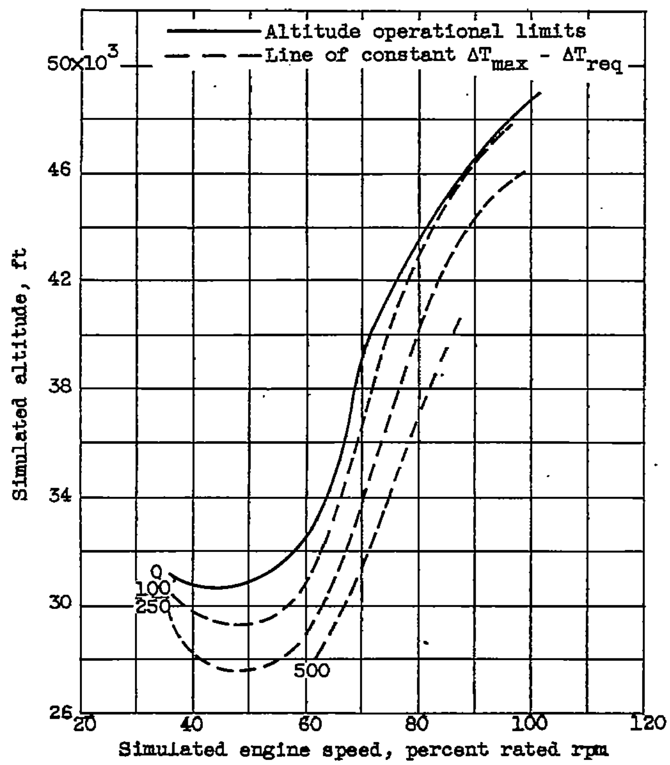


Figure 19. - Available acceleration of turbojet engine as indicated by difference between maximum temperature rise attainable from combustor and temperature rise required for nonaccelerating engine operation. Compressor pressure ratio, 4; flight speed, 0; fuel, AN-F-28.

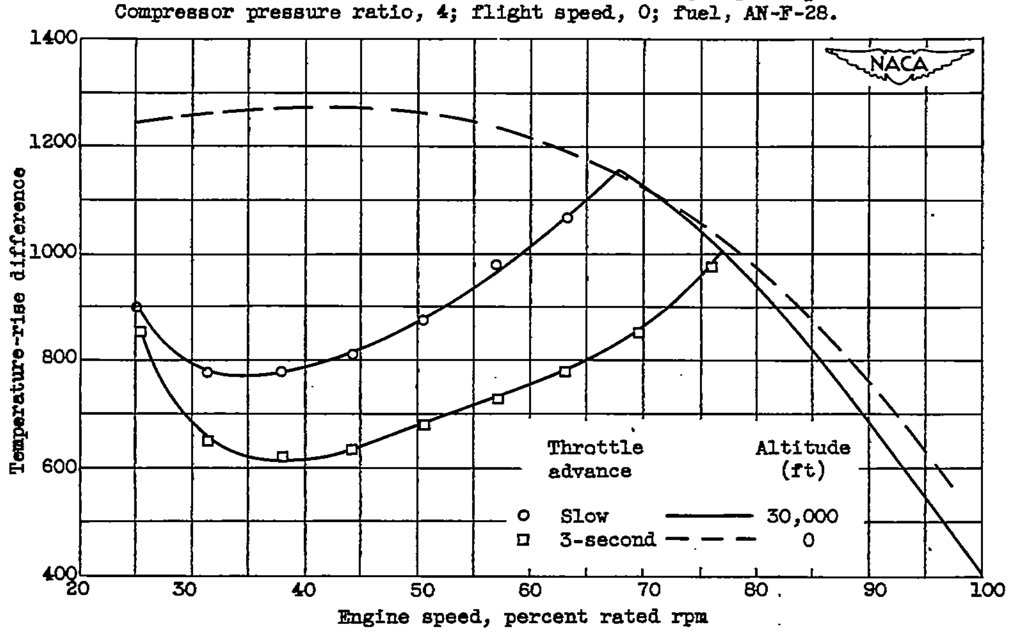


Figure 20. - Excess temperature rise available for engine acceleration. Maximum attainable temperature rise minus temperature rise required for operation. Can-type combustor, fuel, AN-F-48 (weathered); flight Mach number, 0.52.

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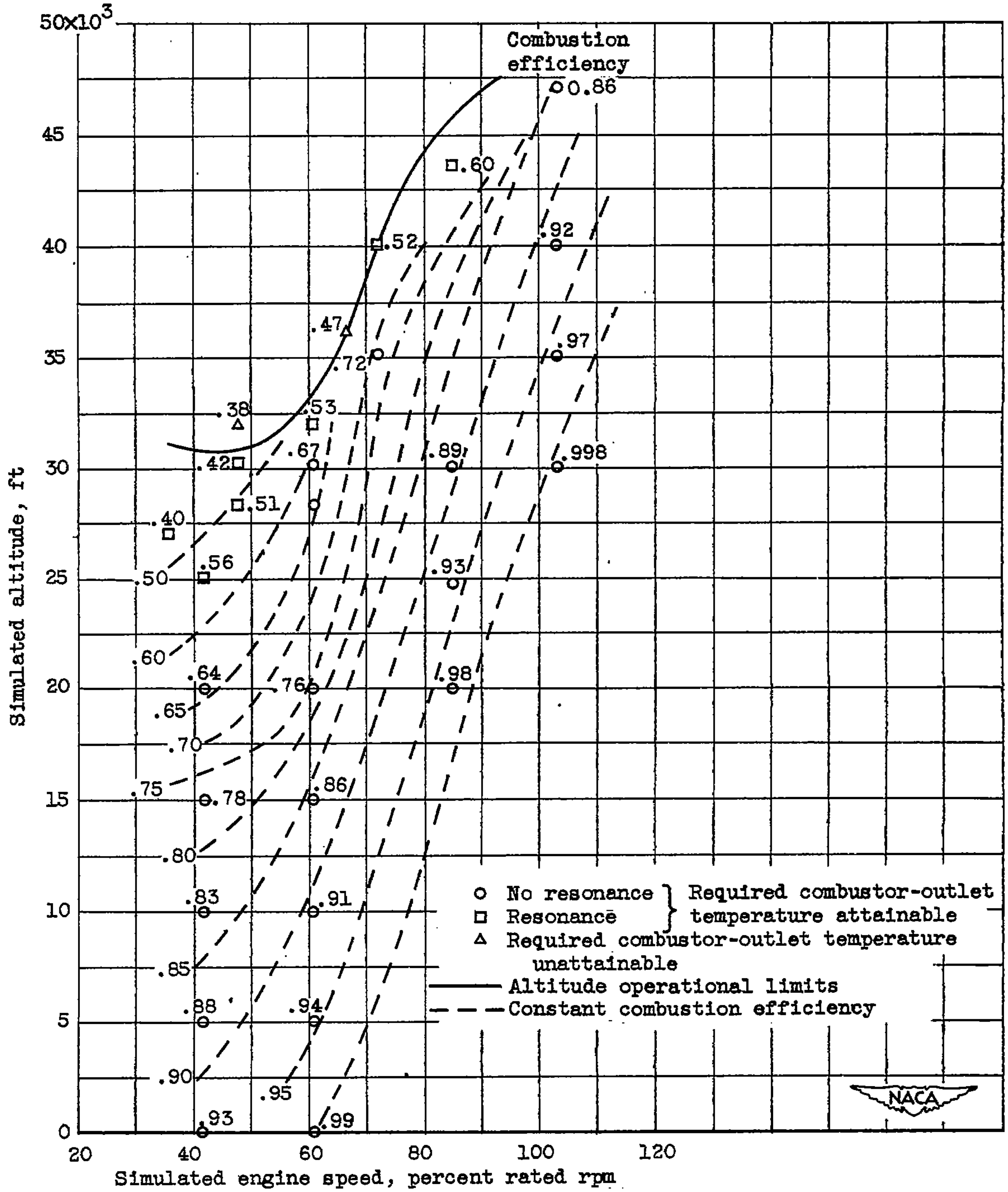

Figure 21. - Combustion efficiencies of typical turbojet combustor at various simulated flight conditions. Compressor pressure ratio, 4; flight speed, 0; fuel, AN-F-28.

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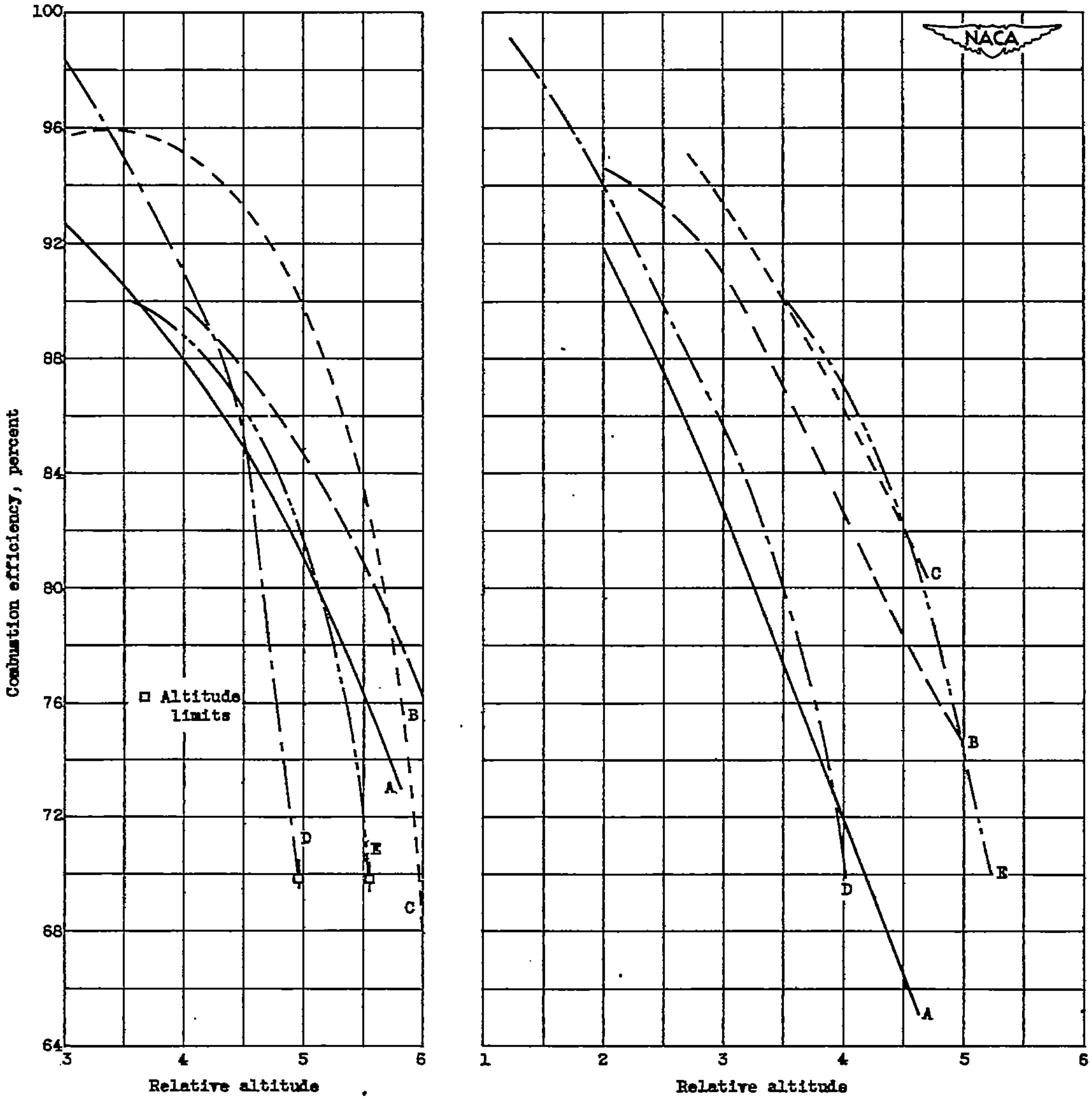

(a) 100-percent rated engine speed.

(b) 80-percent rated engine speed.

Figure 22. - Variation of combustion efficiency with altitude for several turbojet engines as determined from combustor research at simulated conditions for zero flight speed.

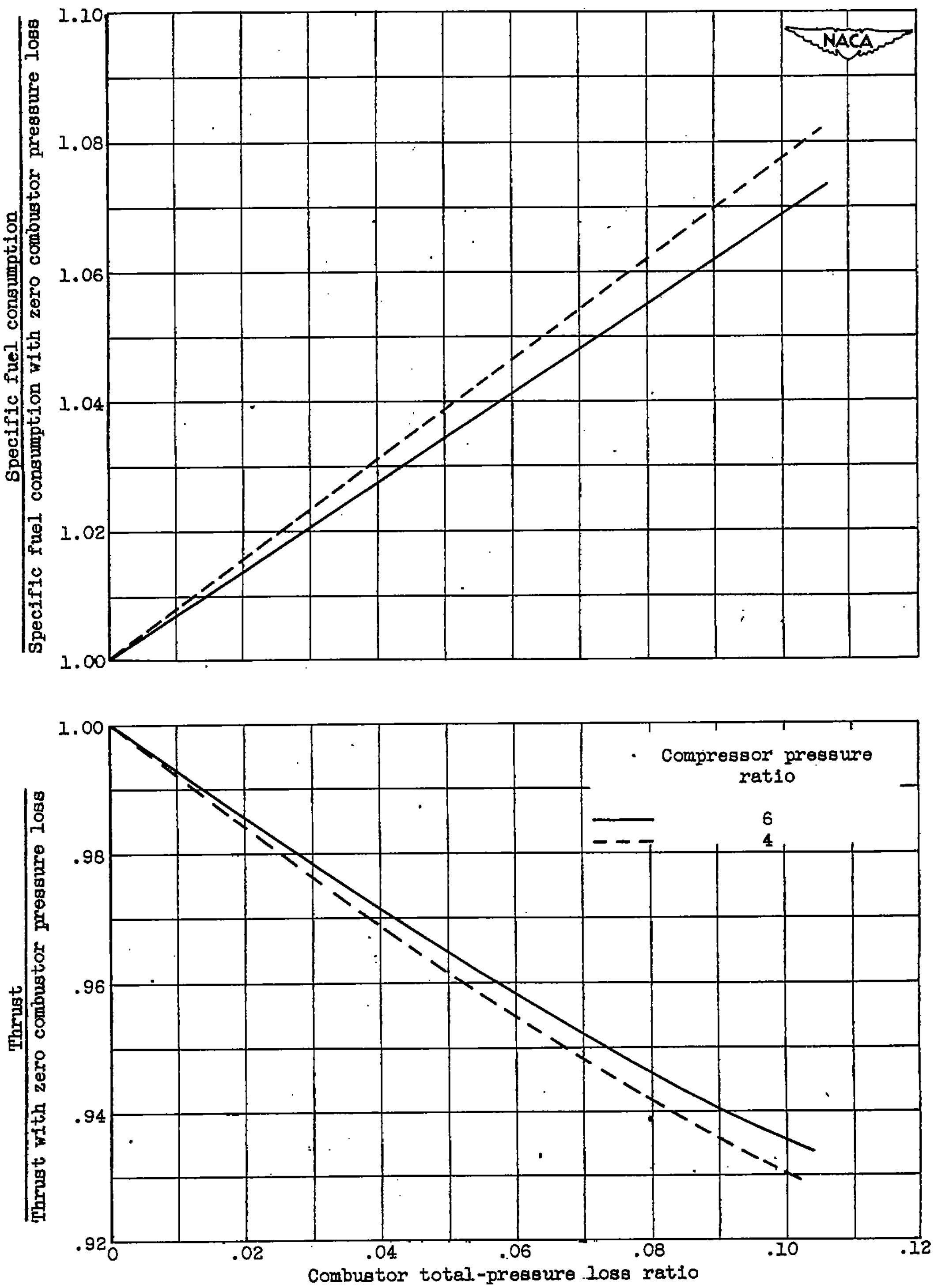

Figure 23. - Effect of combustor total-pressure loss expressed as fraction of total pressure at combustor inlet on thrust and fuel consumption of typical turbojet engine. 100-percent rated rpm; flight Mach number, 0.656; altitude, sea level; turbine-inlet temperature, 1500° F.

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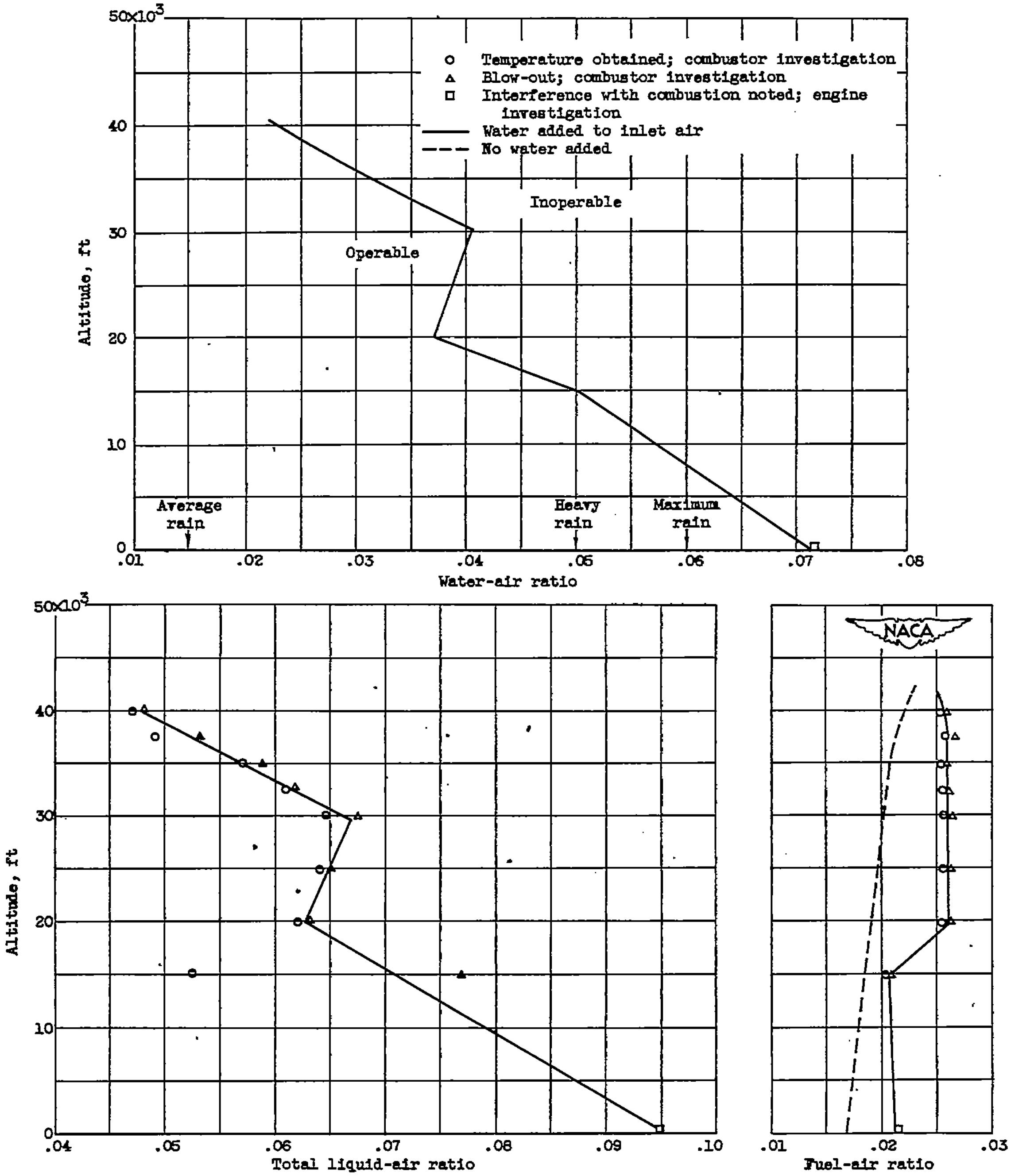

Figure 24. - Maximum permissible water in inlet of turbojet combustor operating at conditions simulating 100-percent rated engine speed and zero flight speed. Water sprayed 62 inches upstream of combustor in inlet duct.

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